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# SXTF FACILITY DESIGN AND DEVELOPMENT SUPPORT

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TRW Defense and Space Systems Group
2340 Alamo SE, Suite 200
Albuquerque, New Mexico 87106

28 May 1981

Final Report for Period 1 January 1980—27 February 1981

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

This document summarizes the effort over the reporting period. The results of site surveys of NASA Houston and Arnold Engineering Development Center are summarized. A study of the possibility of including additional weapons effects testing in the SXTF design is included. A facility-user interface requirements document has been developed and is included. It specifies all of the support required by a satellite manufacturer in order to perform X-ray testing.

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# SECTION 1
# INTRODUCTION

This report summarizes the TRW calender year 1980 effort in support of the development of a Satellite X-ray Test Facility (SXTF). TRW has been involved in the SXTF development program from the early stages as a "surrogate user". The effort has included strawman test planning, assessments of impact and several iterations of facility requirements.[1,2]

The effort for calender year 1980 has included three major areas. First, we have supported the site selection activity. A complete review of the results of our site selection activity is given in Appendix A with a summary in Section 2. Second, we have studied the possibility of including additional weapons effects test capability in SXTF. The result of this study was a briefing which is included as Appendix B. The results are summarized in Section 3. Also, we have formalized the user/facility requirements according to MIL-STD-490 and include these as Appendix C.

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1. Chivington, E.P., et al, "Spacecraft Testing Considerations at SXTF", 34207-6001-UT-00, TRW, March 1980.

2. Chivington, E.P., et al, "Spacecraft Test Planning for a System X-ray Test", 34670-6005-RU-00, TRW, May 30, 1980.

## SECTION 2
## SITE SELECTION

We have evaluated the Arnold Engineering Development Center (AEDC) Mark I and NASA Houston Chamber A candidate SXTF sites, to determine whether there is a preference from a user point of view. The study addresses spacecraft chamber logistics, suitability for alternate uses and chamber size from the point of view of test orientations and thermal control. We have considered spacecraft accommodations primarily in the chamber and have not looked at other areas. Accommodations outside the chamber can be made suitable during the facility modifications.

### 2.1 SPACECRAFT LOGISTICS

In order to determine whether there were major differences between the two candidate sites, a handling sequence was developed. A number of alternatives were explored for each chamber. Some of the normal spacecraft procedures and mechanical support equipment were not suitable so that special adaptations were required. There were no major differences between the facilities for handling the spacecraft. The order of installing spacecraft pieces and the placement of support equipment were different. But, these differences were not significant. The major difference is that the 30-foot diameter AEDC chamber is so restrictive that handling of the FLTSATCOM Satellite and installation of its appendages becomes a hazardous operation. Work platforms, satellite, personnel and support fixtures are positioned in such close proximity inside the chamber that extreme care would be required to avoid injury to the personnel and damage to hardware.

### 2.2 CHAMBER SIZE

The AEDC chamber has a working diameter of about 30 feet versus a working diameter in NASA of about 50 feet. This makes for very cramped quarters for FLTSATCOM. The most likely direction of arrival of x-rays operationally is directly into the antenna reflector. In this configuration, the spacecraft center body will be about 8 meters from the source. Also, because of the cramped quarters, rotating the spacecraft becomes fairly complicated.

The most likely operational exposure direction for DSP is straight up the telescope. This orientation is not possible at AEDC for the newest DSP configuration.

Because space is so limited at AEDC, a test article might have to be quite close to the wall to obtain certain configurations. With a spacecraft panel very close to the wall, the panel temperature will be influenced mostly by the local wall temperature. This may restrict the spacecraft orientations which can be tested because a high heat output panel may not be safe next to a nonshrouded chamber wall area.

## 2.3 ALTERNATE USES

A study was performed to determine whether additional threat simulations could be integrated into the SXTF design. Of the threats (lasers, pellets and EW), lasers seemed to have a number of similar requirements for test configuration. We have looked at the in-chamber laser simulation requirements to determine whether there are any differences between the AEDC and NASA chambers. The approach does not require major alterations to the basic concept, but would require that space be left for integration of laser sources outside the chamber, feedthrough ports and beam handling optics inside the chamber.

# SECTION 3
# ADDITIONAL WEAPONS EFFECTS

SXTF represents a resource for testing spacecraft in x-ray and electron threat environments. Other weapons effects potentially important for spacecraft survivability include lasers, electronic countermeasures (ECM) and pellets. It appeared to us that the basic SXTF concept could be expanded to include additional weapons effects test capability with little impact on the basic function of x-ray and electron testing. Furthermore, there would be a significant cost savings over building separate facilities for the other weapons effects because of the many common features.

The study summarizes the additional threats considered and how they adversely affect spacecraft missions. We summarized the countermeasures and hardening employed and which might require test verification. We also determined whether there were features of the threats which would require system level verification. Of the three threats considered, we concluded that lasers and ECM could require system verification, while pellets probably would not. We also concluded that facility requirements for laser testing, i.e., thermal vacuum chamber and support facilities, were very compatible with SXTF. For ECM some common requirements exist, but there is no requirement for the thermal vacuum chamber. An adjacent anechoic chamber may be more appropriate for the electromagnetic illumination.

X-ray survivability is achieved primarily through hardening of the spacecraft. Methods of hardening for lasers, pellets, ECM and also possibly x-rays may eventually include control from the spacecraft ground station. These hardening methods could consist of avoidance maneuvers and decoy deployment. For these countermeasures ground crew response, recognition of attack type, response time and partial damage assessment are important. Because the ground segment is part of the overall space system, it might be useful to include a simulated ground station at SXTF. This way the survivability of the <u>system</u> could be tested in the presence of simulated threats.

SXTF presents a unique opportunity to provide system level test verification for x-rays. An SXTF requirement, however, also presents a major investment in resources to move, setup, test and return a spacecraft. It also requires a major investment in facility modifications. A natural expansion of the proposed capability is to provide a single resource for all system level weapons effects test verification.

Much of the spacecraft preparation, setup and test equipment then becomes common among x-rays, lasers and ECM.

The study results are included in Appendix B and are given in show-and-tell format.

## SECTION 4

## FACILITY INTERFACE REQUIREMENTS

Much of the effort this past year has been to update and formalize facility requirements from a user point of view. This has been completed and is included in Appendix C . These requirements have been updated to reflect the choice of the AEDC Mark I Chamber. The requirements also reflect an effort to make minimal modifications to existing AEDC facilities while still ensuring user compatibility. The document specifies what interfaces exist between the user and the facility, where those interfaces should be located and the physical and functional configuration of those interfaces. The document also includes size, power and weight requirements. The requirements follow the format specified in MIL-STD-490 for facility specifications.

Some specific areas in the requirements which are of particular impact are included here for emphasis.

### VESTIBULE AREA

We have specified a vestibule area capable of handling a 32 ft. x 12 ft. spacecraft transporter. This includes removing the cover and lifting the spacecraft up to the satellite buildup area.

### USER SCREEN ROOM

We have required a screen room of at least 1350 $ft.^2$. This is larger than has been requested in the past but is based on careful layout of the room including the equipment involved.

### POWER/CHARGING UMBILICAL

A method is required to prevent chamber transients from entering the screen room on the power cable. This can either be done with sufficient shielding of the power cable or with a disconnect at the user screen room.

The interface for the power umbilical is still not entirely settled. The alternatives include a user supplied receptacle on the spacecraft which mates with a standard facility cable and retraction mechanism. This has limitations because of the various places that a spacecraft receptacle might exist and because of the spacecraft unique powering interface. Another alternative would be for the user to supply all hardware from the chamber feedthrough to the spacecraft. This would be an expensive burden for each user. An acceptable compromise may be to require the user to supply the chamber to spacecraft umbilical for current spacecraft and require that new spacecraft incorporate provisions for a standardized interface compatible with a facility provided umbilical.

## CHAMBER THERMAL CONTROL

The existing AEDC thermal control system could be used for spacecraft testing if zonal control is retained. A special thermal shield will be required over the sources. Special equipment such as on-board heaters, thermal shields and chamber mounted light sources may be required to protect certain spacecraft hardware. These may degrade the quality of the x-ray and electron charging experiments. An alternative is to modify the facility to provide gaseous nitrogen cold walls with minimal zonal controls.

A requirement exists to protect the spacecraft from very cold temperatures if the electronics are not powered. During a facility power outage this could occur if the cold walls could not be brought up to ambient in a few hours and the power umbilical could not be reinserted to power the spacecraft. Our approach to this has been to require the cold walls to heat up to ambient in two hours which is about the time that the spacecraft would run out of battery power.

## CHAMBER SUSPENSION

Suspension requirements are based on the preliminary conclusion that support from above will be the preferred method. Further studies may be appropriate to verify this conclusion. The reason that the overhead suspension appears to be preferrable is that it minimizes the amount and complexity of extraneous material in the test volume. Even dielectric material can interfere with the photon experiment and will certainly interfere with spacecraft charging experiments. The amount of material to support the test object weight as well as the fixture's own weight will not be insignificant. Furthermore, a structure capable of supporting the spacecraft from below will have to have a large enough base to prevent toppling.

Considering spacecraft like FLTSATCOM and DSCS-III, the fixture would have to have a large open area directly below the center body for the lower solar panel and a lattice work above the spacecraft for the upper solar panel. It was because of these configurations that we reached our preliminary conclusion.

There are three major problems associated with the dielectric suspension mount:

1) The number of lines and attach points are unique to each spacecraft requiring a significant user design effort,

2) the facility provided strongback would be modified (holes drilled, etc.) by each user to accommodate each unique requirement thus limiting the strongback's total useful lifetime, and

3) the transition involved in transferring the strongback from the facility crane to the rotation fixture will be a difficult technical operation requiring utmost care. Aside from these problems, the suspension mount is considered the best method for holding and rotating the spacecraft.

# APPENDIX A

## EVALUATION OF CANDIDATE SXTF SITES FOR USER COMPATIBILITY

# OVERVIEW

We have evaluated the AEDC and NASA candidate sites to determine whether there is a preference from a user point of view. We have considered spacecraft accommodations primarily in the chamber and have not yet looked at other areas. Outside the chamber, accommodations can be made suitable during the facility modifications. The conclusions of our study are:

- Setup operations will be very cramped at AEDC increasing the risk of damage to spacecraft and injury to personnel.
- The time required for setup and the amount of special test equipment (MAGE) are probably about the same at the two facilities.
- The preferred orientation of the new DSP satellite cannot be tested at AEDC.
- Laser illumination of the FLTSATCOM and DSP can be performed at both facilities.
- Security at NASA will be more difficult to implement because all of the secure areas are not contiguous, and because the facility is basically unsecured.
- Spacecraft test orientations may be restricted at AEDC due to thermal control problems resulting from the spacecraft having to be very close to the chamber walls.

## DISCUSSION

### SPACECRAFT LOGISTICS

In order to determine whether there are major differences between the two candidate sites, we have developed a spacecraft handling sequence. We have a step by step procedure for unpacking, preparing, installing and suspending the spacecraft. The ground rules for the installation were as follows:

- Spacecraft to be suspended by dielectric ropes with the long axis of the spacecraft parallel to the cylinder axis.
- Spacecraft to have a functional check prior to test.
- Use existing ground support equipment (GSE) and off the shelf items, wherever possible.
- Special (SGEMP) instrumentation installed in plant prior to shipping to SXTF.

A number of alternatives were explored for each of the chambers. Some of the normal spacecraft procedures and mechanical support equipment were not suitable for the setup. The procedures finally developed are outlined in Table A-1 for the two facilities. The steps are shown side by side to help compare the two. Figures A-1 through A-11 show key steps in the procedures for the AEDC chamber and Figures A-12 through A-26 for the NASA chamber. Table A-2 is a list of test support equipment requirements for the chamber.

The major difference between the two procedures is the order in which steps are accomplished. Space is very limited in the AEDC chamber. Therefore, only one of the solar panels is placed inside the chamber before beginning spacecraft buildup. Furthermore, in AEDC great care must be taken in the placement of support equipment to allow for parallel activities. This is particularly apparent in Figures A-8, 9 and 10.

The procedures for AEDC were developed first. Because equipment cannot be removed through the main door after the spacecraft is installed, all hardware must fit through the 8-foot door. Once an acceptable procedure was developed for AEDC, the same was used for NASA so that ultimately the chamber door size did not appear to make a difference. The 30-foot diameter AEDC chamber is so restrictive that handling of the FSC Satellite and installation of its appendages becomes a very hazardous operation. Work platforms, satellite, personnel and GSE support fixtures are positioned in such close proximity inside the chamber that extreme care would be required to avoid injury to the personnel and damage to hardware.

Another difference is the use of the overhead fixture. At AEDC, a separate overhead crane is required to place equipment in the chamber, and the support fixture must be moved back out of the way. At NASA the equipment is placed in the chamber using a stinger crane, then the support fixture is used for all of the lifting.

CHAMBER SIZE

The AEDC chamber has a working diameter of about 30 feet versus a working diameter in NASA of about 50 feet. As shown in Figure A-27, this makes for very cramped quarters for FLTSATCOM. The most likely direction of arrival of x-rays operationally is directly into the antenna reflector. In this configuration, the spacecraft center body will be about 8 meters from the source. Also, because of the cramped quarters, rotating the spacecraft becomes fairly complicated. Figure A-28 shows the path of the spacecraft center of gravity to rotate 90°. Figures A-29 through A-32 show various other orientations of FLTSATCOM in the AEDC chamber.

The most likely operational exposure direction for DSP is straight up the telescope. As one can see from Figure A-33, this orientation is impossible at AEDC.

ALTERNATE USES

A study was performed to determine whether additional threat simulations could be integrated into the SXTF design. Of the threats (laser, pellets and EW), lasers seemed to have a number of similar requirements for test configuration. We have looked at the in-chamber laser simulation requirements to determine whether there are any differences between the AEDC and NASA chambers. Laser testing might employ two approaches to illuminating the spacecraft. Tests requiring particular spectral content would require a laser outside the chamber with a small window and beam handling optics (BHO) inside the chamber. Figures A-34 and A-35 show a scaled representation of the AEDC and NASA laser integrations. A possibly more interesting orientation for FLTSATCOM is shown in Figure A-36 for the NASA chamber. The orientation of FLTSATCOM will not work at AEDC however, a vertical orientation may work if the beam handling optics can fit in between the spacecraft and the walls.

Tests requiring a simulation of the laser heat input may be most cost effectively done with a Cassegrain quartz halogen lamp bank. This system can be integrated into AEDC without great difficulty as shown in Figure A-37.

## THERMAL CONTROL

Because space is so limited at AEDC, the spacecraft might have to be quite close to the wall to obtain certain configurations. With a spacecraft panel very close to the wall, the panel temperature will be strongly influenced only by the local wall temperature. This may restrict the spacecraft orientations which can be tested because a high heat output panel may not be safe next to a nonshrouded chamber wall area.

Table A-1 Satellite Test Preparation Sequence

| | AEDC Chamber | | Houston Chamber |
|---|---|---|---|
| 1. | Deliver test equipment, validate and prepare site for receipt of satellite<br>• prepare floor of chamber to provide stable work platform<br>• lower specified MAGE & support equipment through the top of chamber onto the floor<br>• place specified test racks adjacent to chamber at ground level | 1. | Deliver test equipment, validate and prepare site for receipt of satellite<br>• prepare floor of chamber to provide stable work platform<br>• install specified MAGE & support equipment through 40 ft dia door and position on floor of chamber<br>• place specified test racks adjacent to chamber at ground level |
| 2. | Transport to ADEC via C5A | 2. | Transport to JSC via C5A |
| 3. | Transport satellite via road transport to test facility | 3. | Transport satellite via road transport to test facility |
| 4. | Outside test facility, remove satellite transporter cover | 4. | Move satellite transporter into airlock and clean transporter prior to moving into the high-bay |
| 5. | Move satellite transporter into chamber high-bay area | 5. | Move satellite transporter into chamber high-bay area |
| 6. | Hoist satellite into satellite prep area, above chamber, and set it down on prepositioned pedestal | 6. | Remove satellite transporter cover (Figure A-12) |
| 7. | Remove protective covers from satellite | 7. | Hoist satellite out of transporter and position on preposition pedestal (Figures A-13, 14 and 15) |
| 8. | Remove solar arrays from satellite and install on strongback dolly | 8. | Remove protective covers from satellite |
| 9. | Remove helix portion of receive antenna and place in storage container | 9. | Position satellite rotation fixture in chamber (Figure A-16) |
| 10. | Move the helix portion of the receive antenna to the room adjacent to the 8' dia door | 10. | Remove solar arrays from satellite and install on strongback dolly (Figure A-17 and 18) |
| 11. | Lower one solar array into chamber (Figure A-1) | 11. | Remove helix portion of receive antenna and place in storage container |

Table A-1 Satellite Test Preparation Sequence (con't)

| AEDC Chamber | Houston Chamber |
|---|---|
| 12. Position satellite rotation fixture in chamber (Figure A-2) | 12. Install satellite on rotation fixture in vertical position (Figure A-19) |
| 13. Install satellite on rotation fixture in vertical position (Figures A-3 and A-4) | 13. Install both solar arrays in chamber adjacent to satellite (Figure A-20) |
| 14. Perform satellite functional test | 14. Perform satellite functional test |
| 15. Rotate solar array booms, receive antenna boom in prep for deployment of transient antenna | 15. Rotate solar array booms, receive antenna boom in prep for deployment of transient antenna |
| 16. Deploy transmit antenna | 16. Deploy transmit antenna |
| 17. Rotate satellite to horizontal position (Figure A-6) | 17. Rotate satellite to horizontal position (Figure A-21) |
| 18. Rotate upper solar array boom to its deployed position (Figure A-7) | 18. Rotate upper solar array boom to its deployed position |
| 19. Install upper solar array | 19. Install upper solar array |
| 20. Attach tether lines to satellite and appendages (Figure A-8) | 20. Attach tether lines to satellite and appendages (Figure A-22) |
| 21. Raise satellite in preparation for installation of lower solar array (Figure A-9) | 21. Install helix portion of receive antenna and tether (Figure A-23) |
| 22. Install lower solar array and tether (Figure A-10) | 22. Raise satellite in preparation for installation of lower solar array (Figure A-24) |
| 23. Position satellite, by adjusting suspension fixture, to prepare for installation of helix portion of receive antenna | 23. Install lower solar array and tether (Figure A-25) |
| 24. Install helix portion of receive antenna and tether | 24. Remove all MAGE from chamber 40' dia door and store adjacent to chamber |
| 25. Remove all MAGE from chamber through 8' dia door and store adjacent to chamber | 25. Position satellite in test location (Figure A-26) |

Table A-1 Satellite Test Preparation Sequence (con't)

| AEDC Chamber | Houston Chamber |
| --- | --- |
| 26. Install chamber lid | 26. Prepare for pump down |
| 27. Position satellite in test location (Figure A-11) | |
| 28. Prepare for pump down | |

Table A-2 MAGE Equipment List

| | |
|---|---|
| Pedestals (3) | G319444 |
| Transporter Adapter Assembly | G316620 |
| Sling Assembly | G311358 |
| Sling Assembly | G317168 |
| Pedestal Table Assembly | G273963 |
| Aft Support Adapter | G273959 |
| Separation Band | G273960 |
| Sling Assembly | G273958 |
| Rotation Fixture | G273961 |
| Sling Assembly for Handling (Rotation Fixture) | TBD |
| Stinger Crane (Rental Unit) | TBD |
| Solar Array Strongback | TBD |
| Sling Assembly for S/A | TBD |
| Tether Lines | TBD |
| Suspension fixture | TBD |

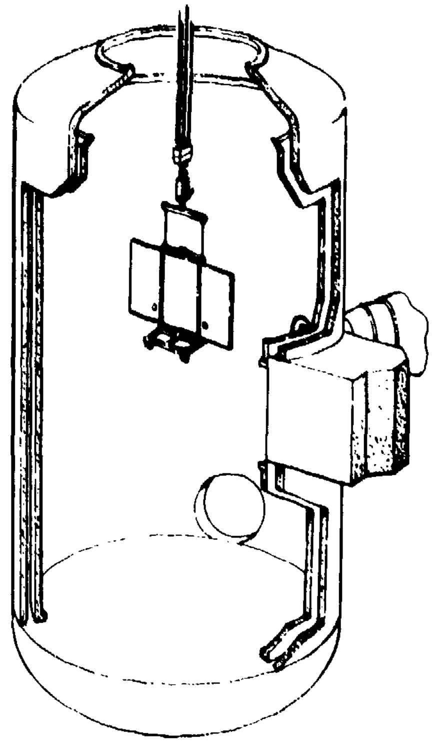


Figure A-1 AEDC Loading Procedure

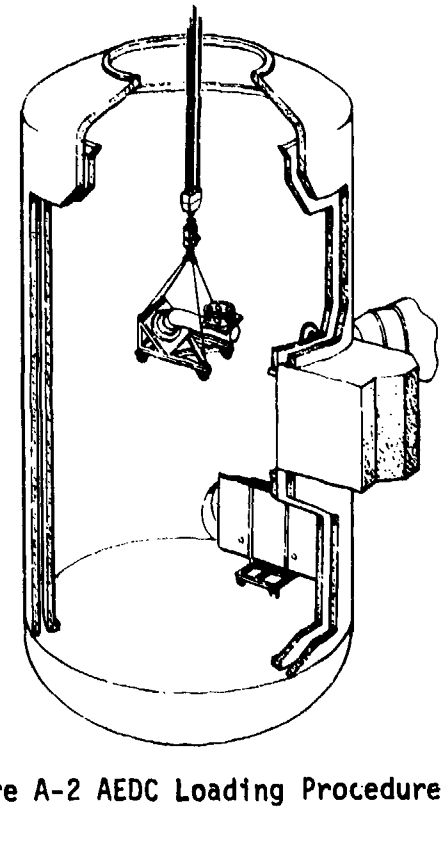


Figure A-2 AEDC Loading Procedure

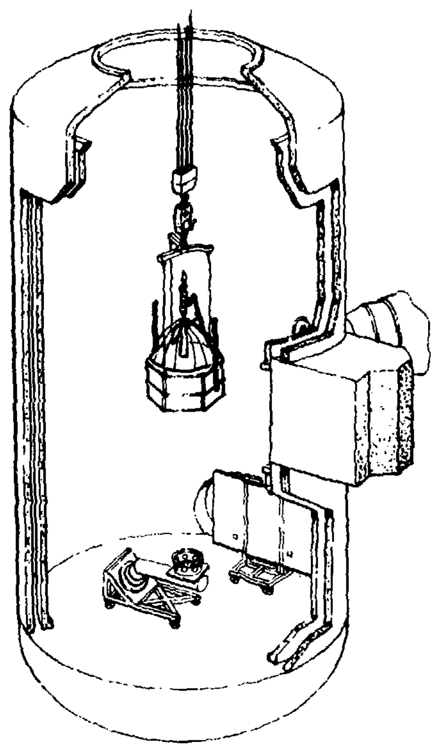


Figure A-3 AEDC Loading Procedure

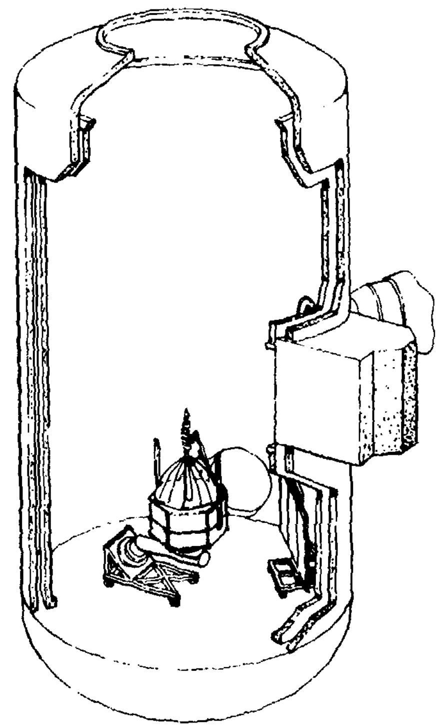


Figure A-4 AEDC Loading Procedure

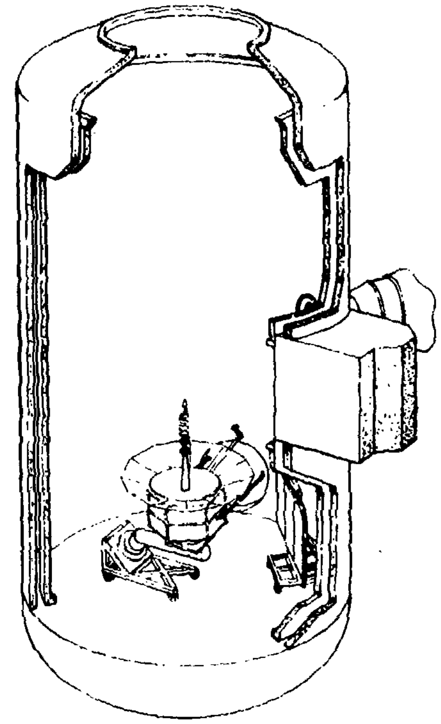


Figure A-5 AEDC Loading Procedure

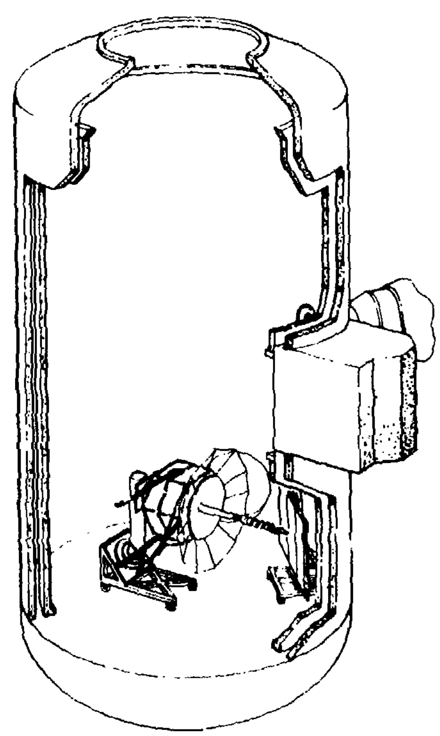


Figure A-6 AEDC Loading Procedure

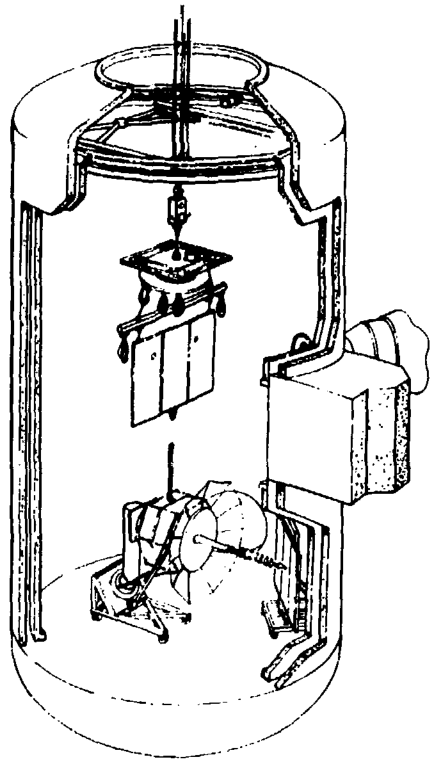


Figure A-7 AEDC Loading Procedure

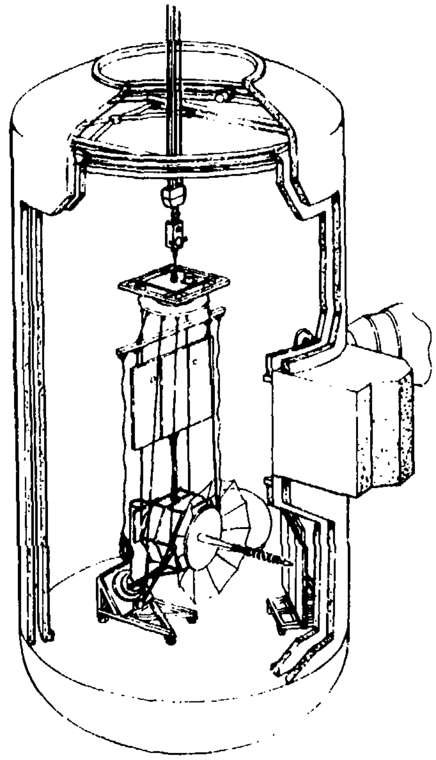


Figure A-8 AEDC Loading Procedure

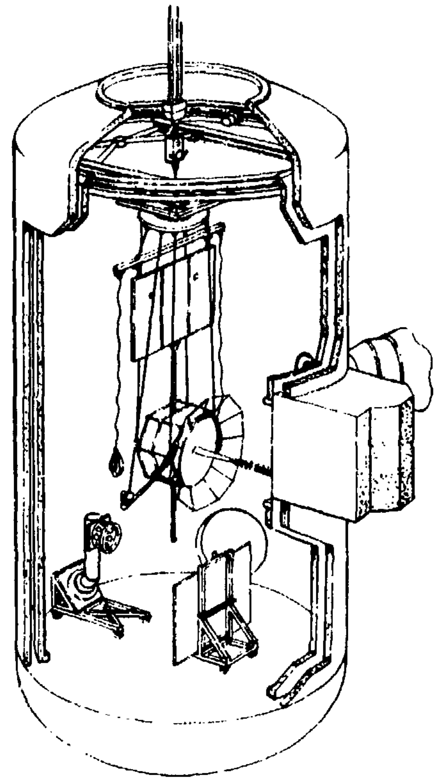


Figure A-9 AEDC Loading Procedure

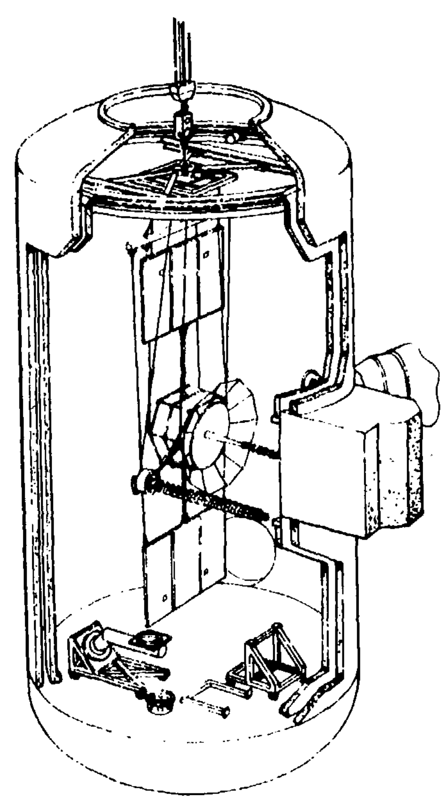


Figure A-10 AEDC Loading Procedure

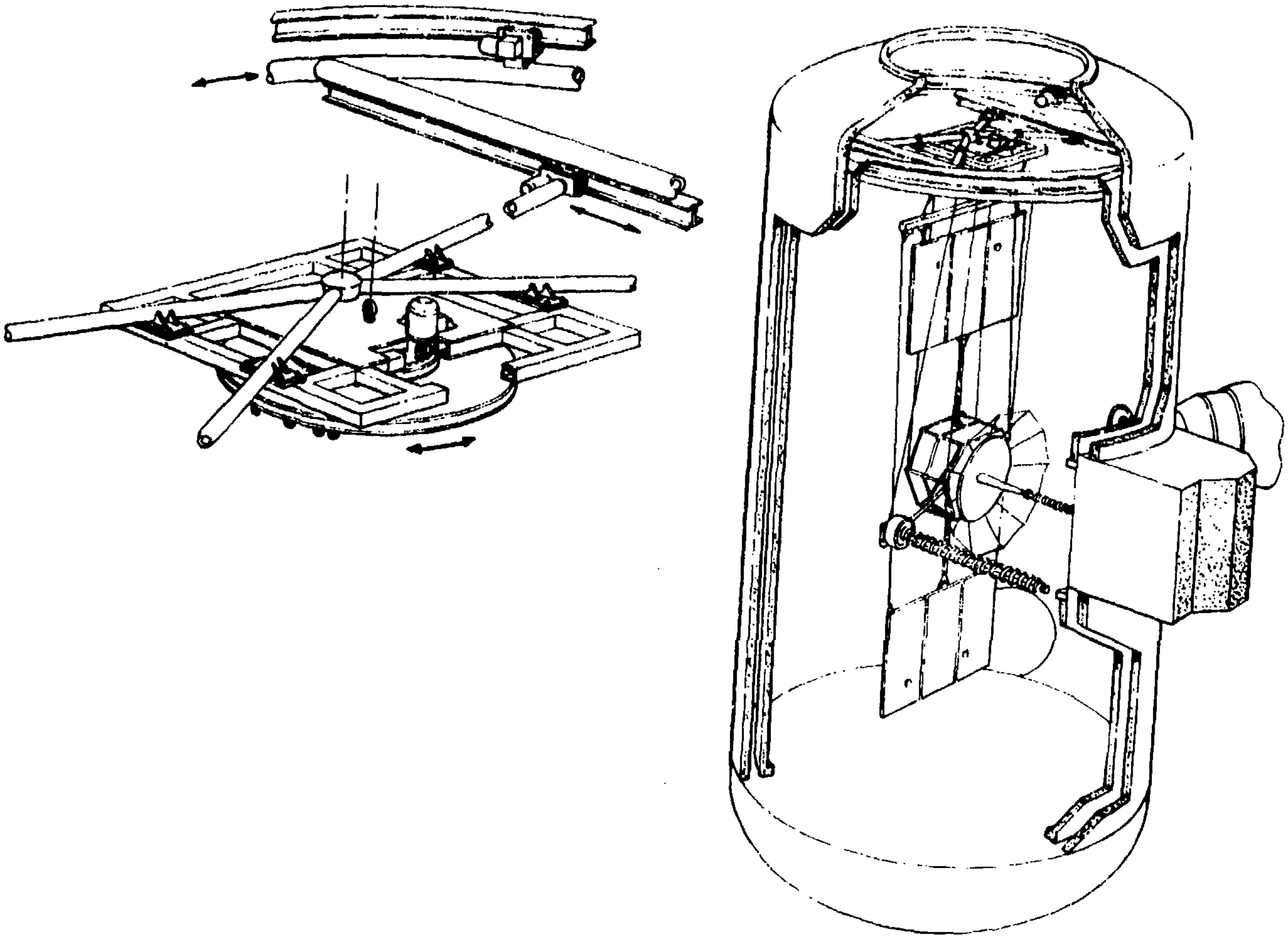


Figure A-11 AEDC Loading Procedure

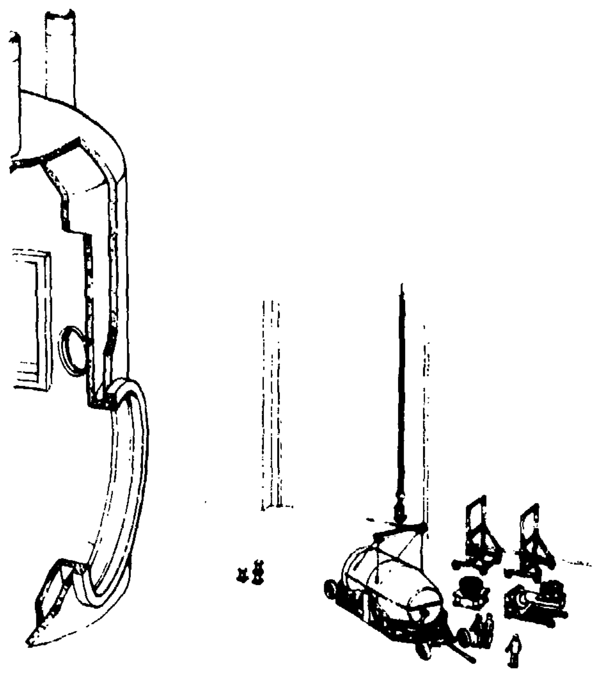


Figure A-12 NASA Houston Loading Procedure

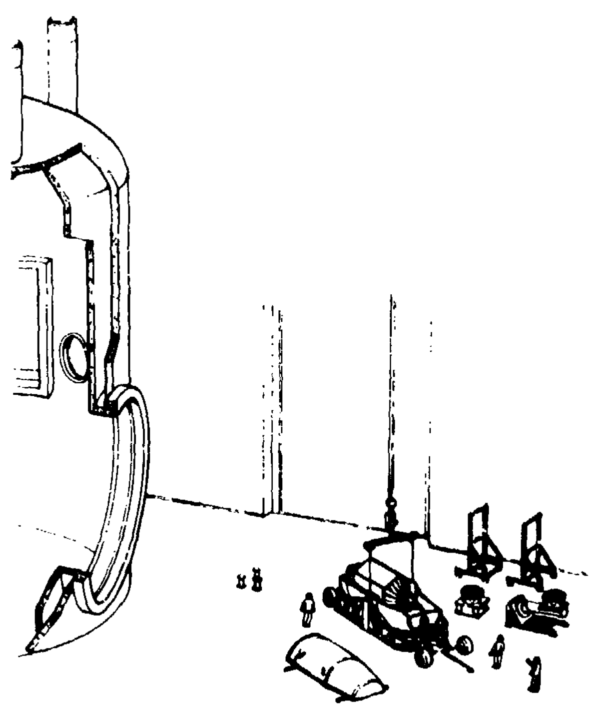


Figure A-13 NASA Houston Loading Procedure

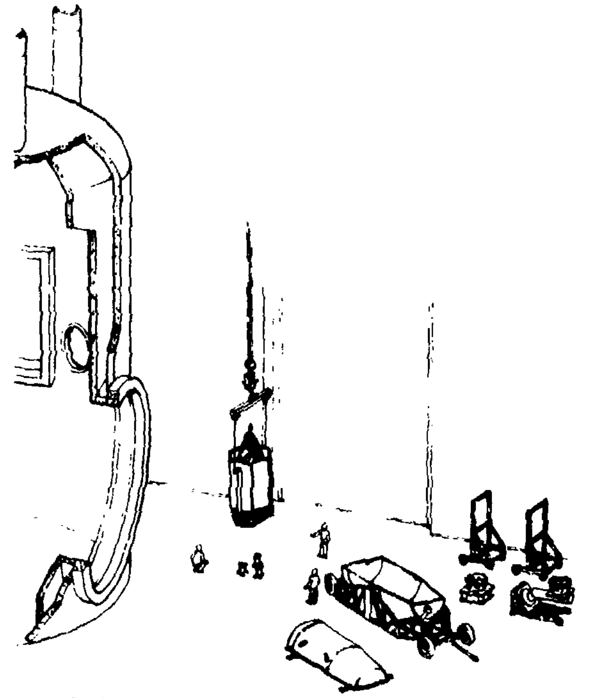


Figure A-14 NASA Houston Loading Procedure

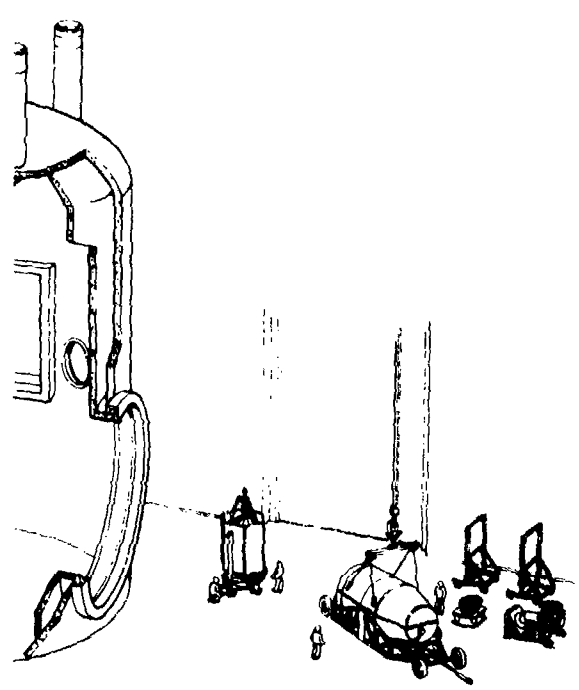


Figure A-15 NASA Houston Loading Procedure

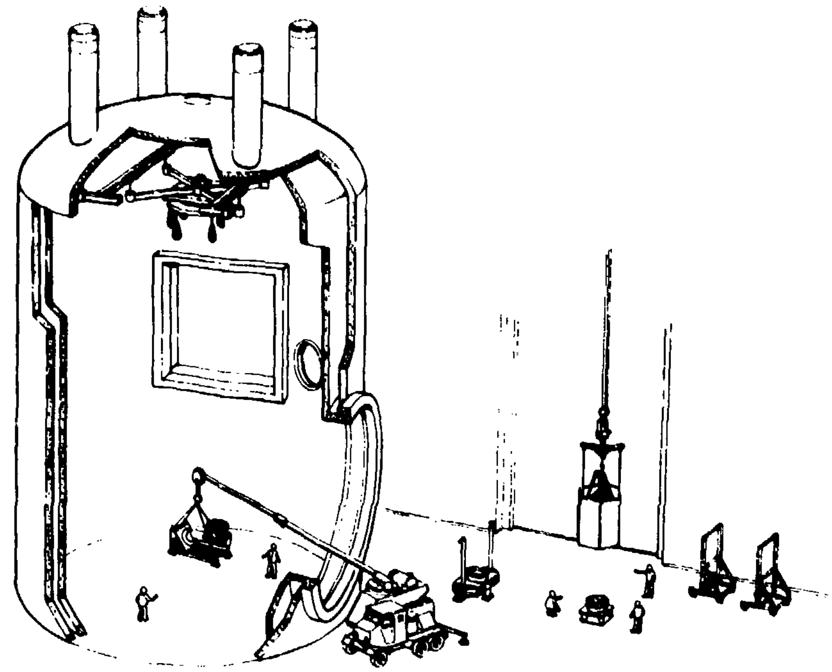


Figure A-16 NASA Houston Loading Procedure

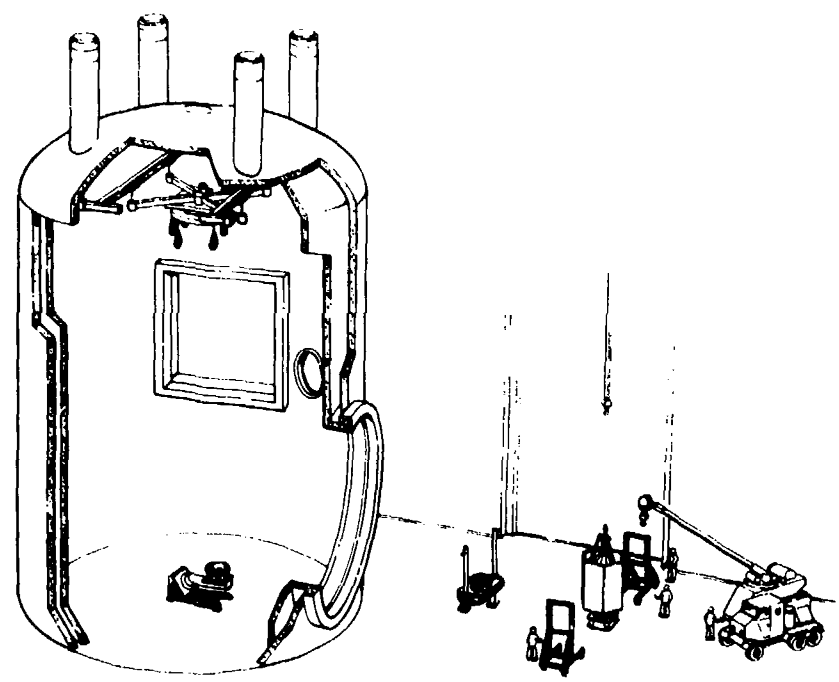


Figure A-17 NASA Houston Loading Procedure

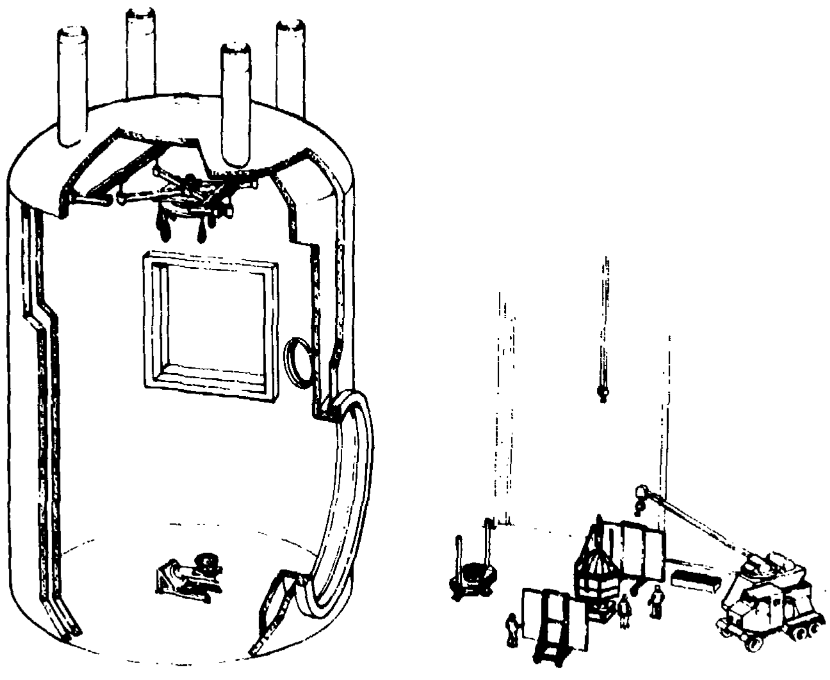


Figure A-18 NASA Houston Loading Procedure

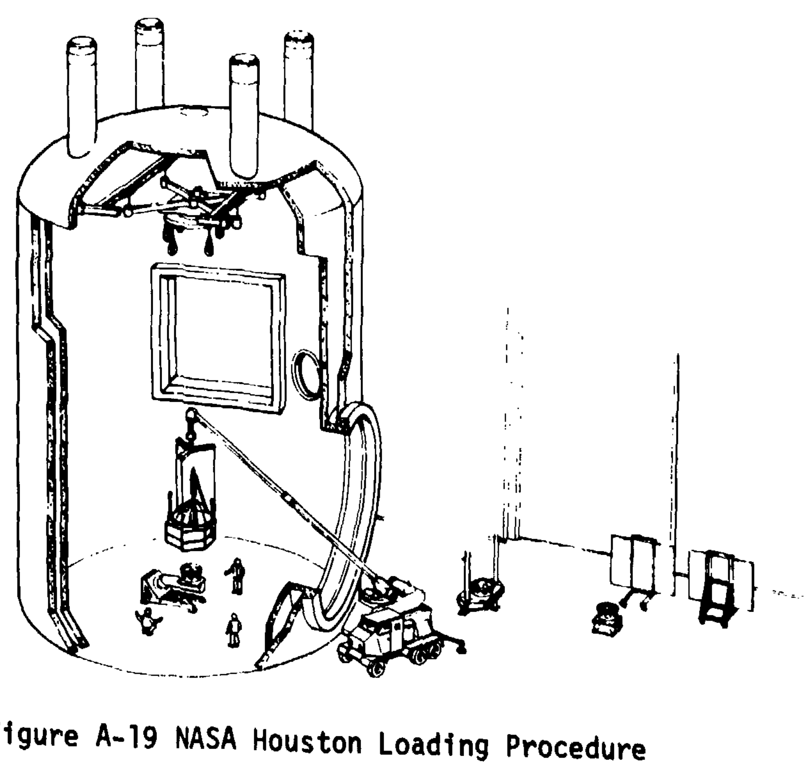
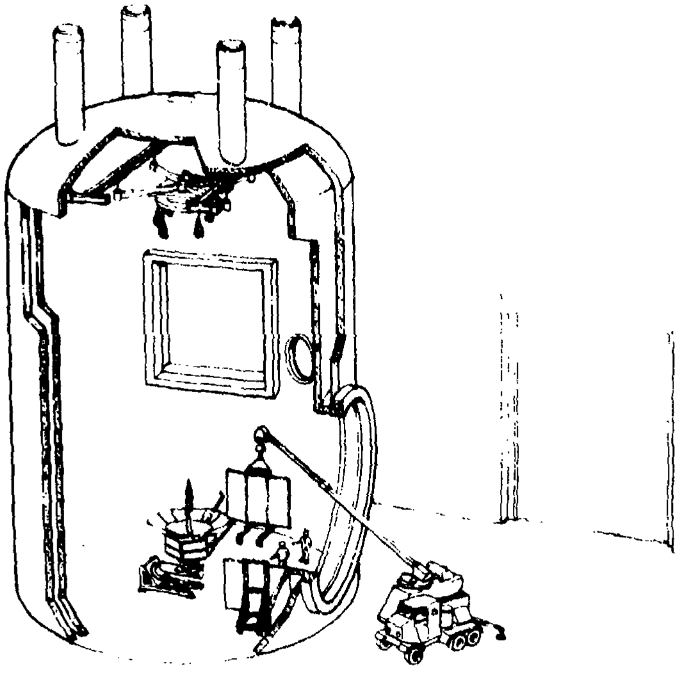


Figure A-19 NASA Houston Loading Procedure



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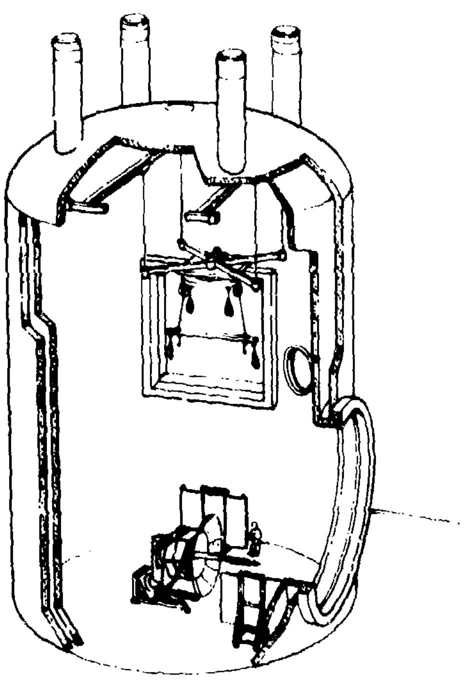

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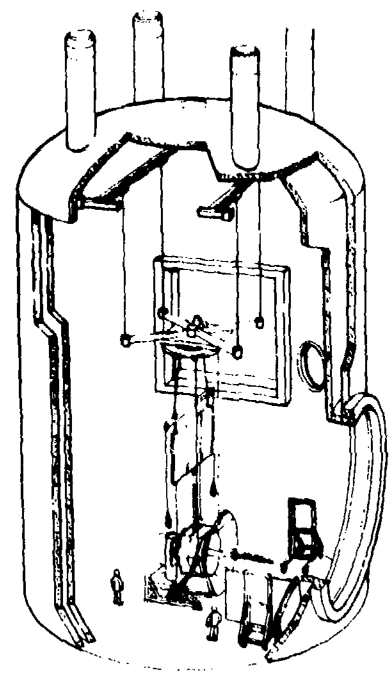
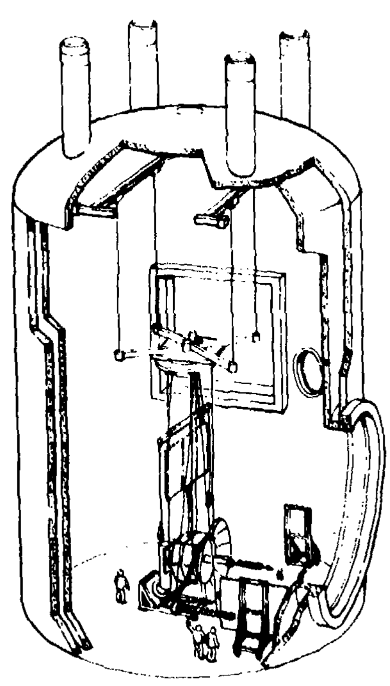


Figure A-22 NASA Houston Loading Procedure



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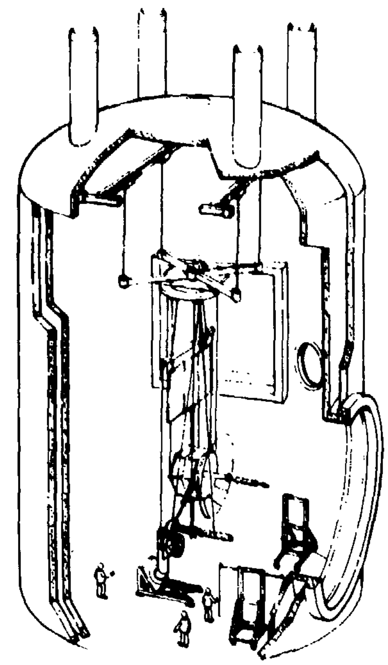

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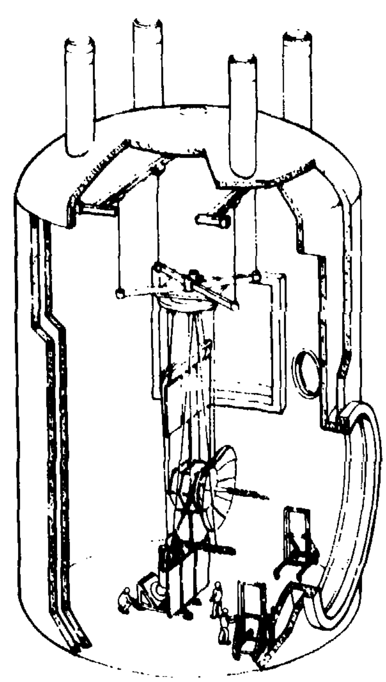

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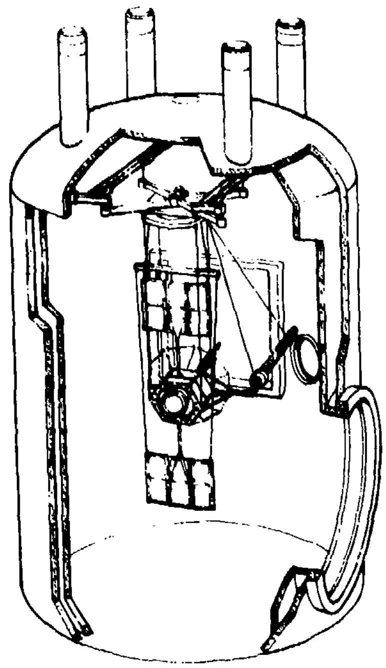

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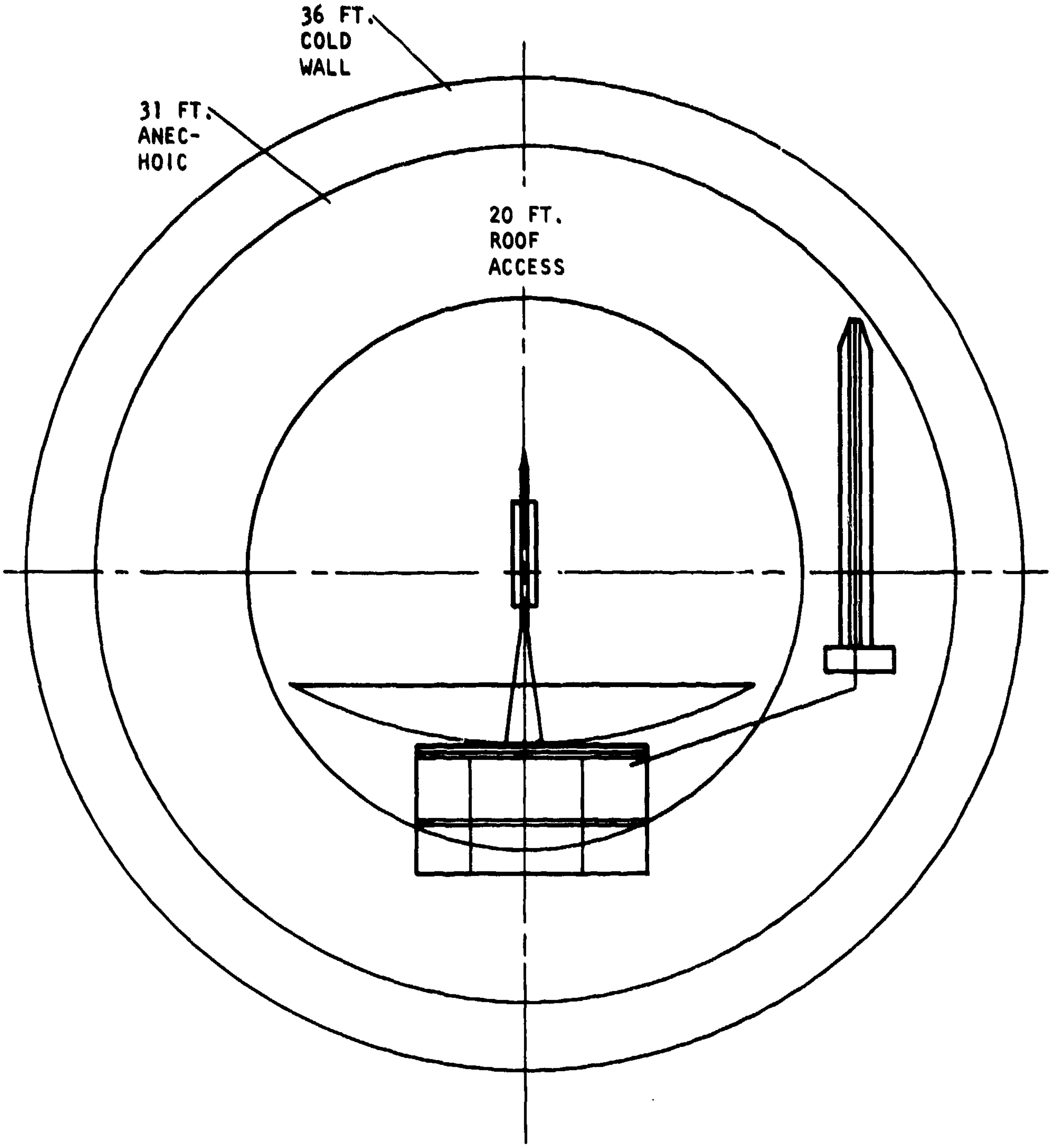



Figure A-27. Spacecraft facing source 24' from source to center body.

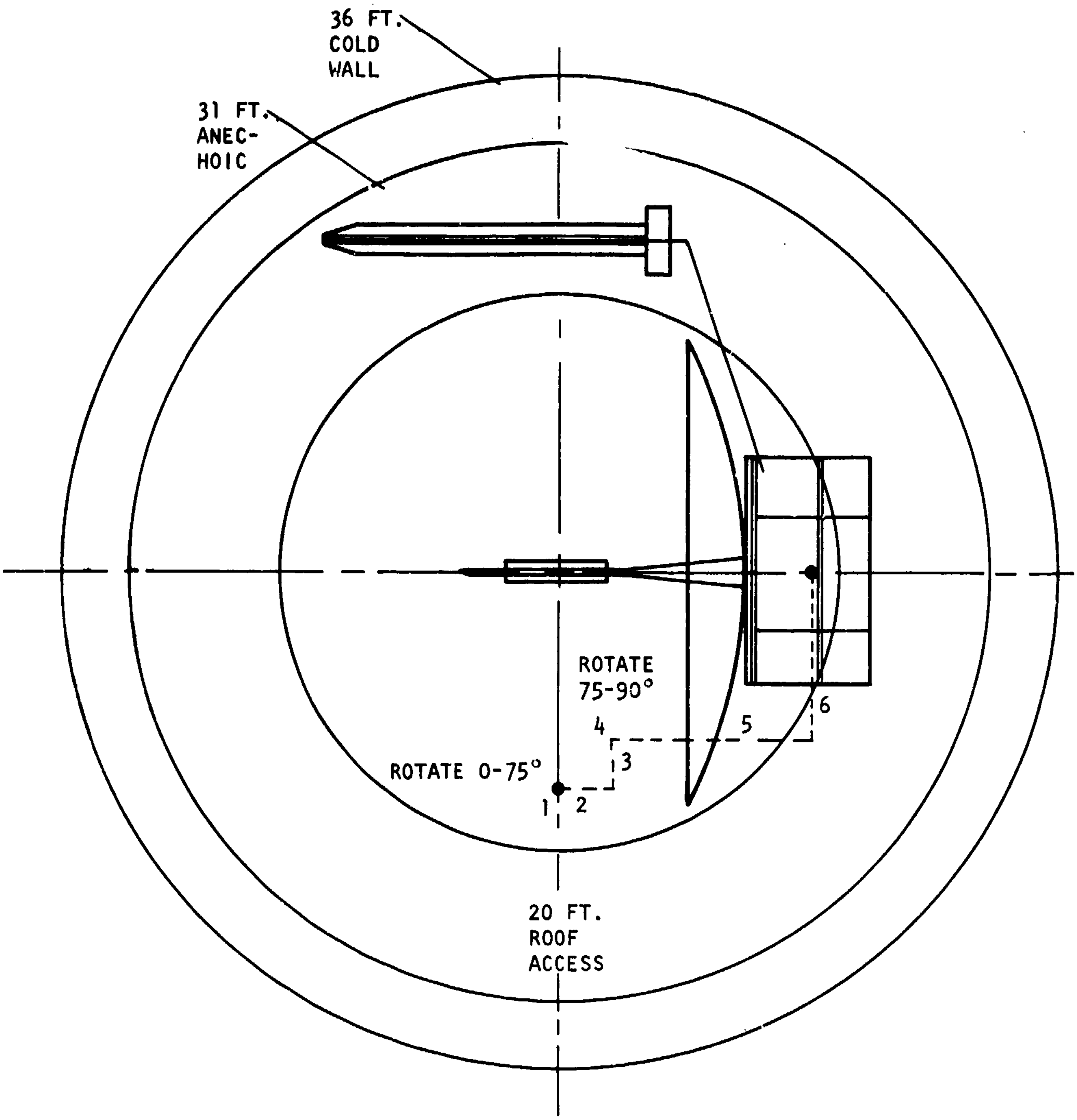



Figure A-28. Spacecraft 90° to source, receive antenna close, spacecraft source axis.

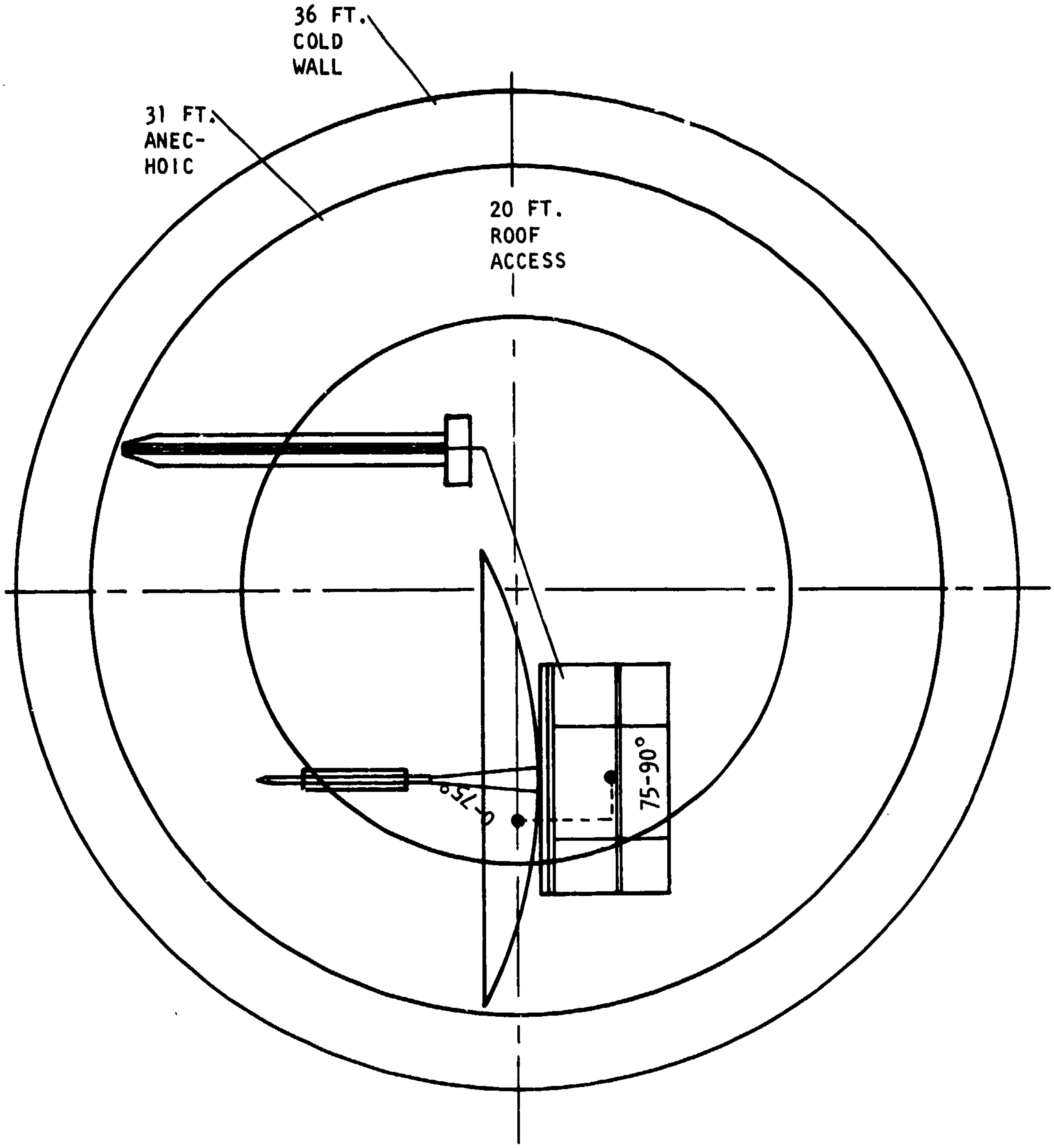



Figure A-29. Spacecraft 90° to source, receive antenna close, spacecraft near center.

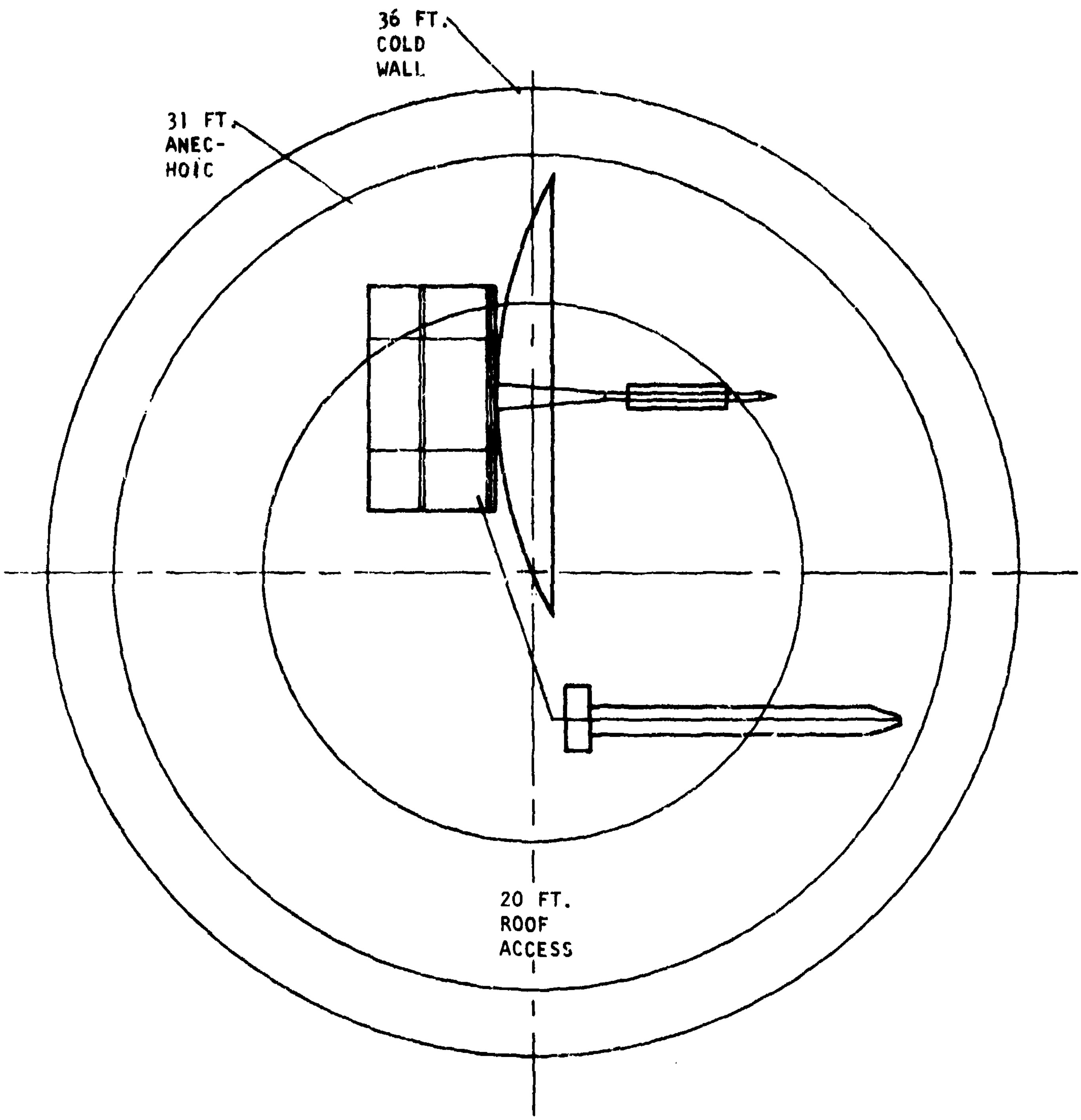



Figure A-30. Spacecraft 90° to source, receive antenna away, center body close to source axis.

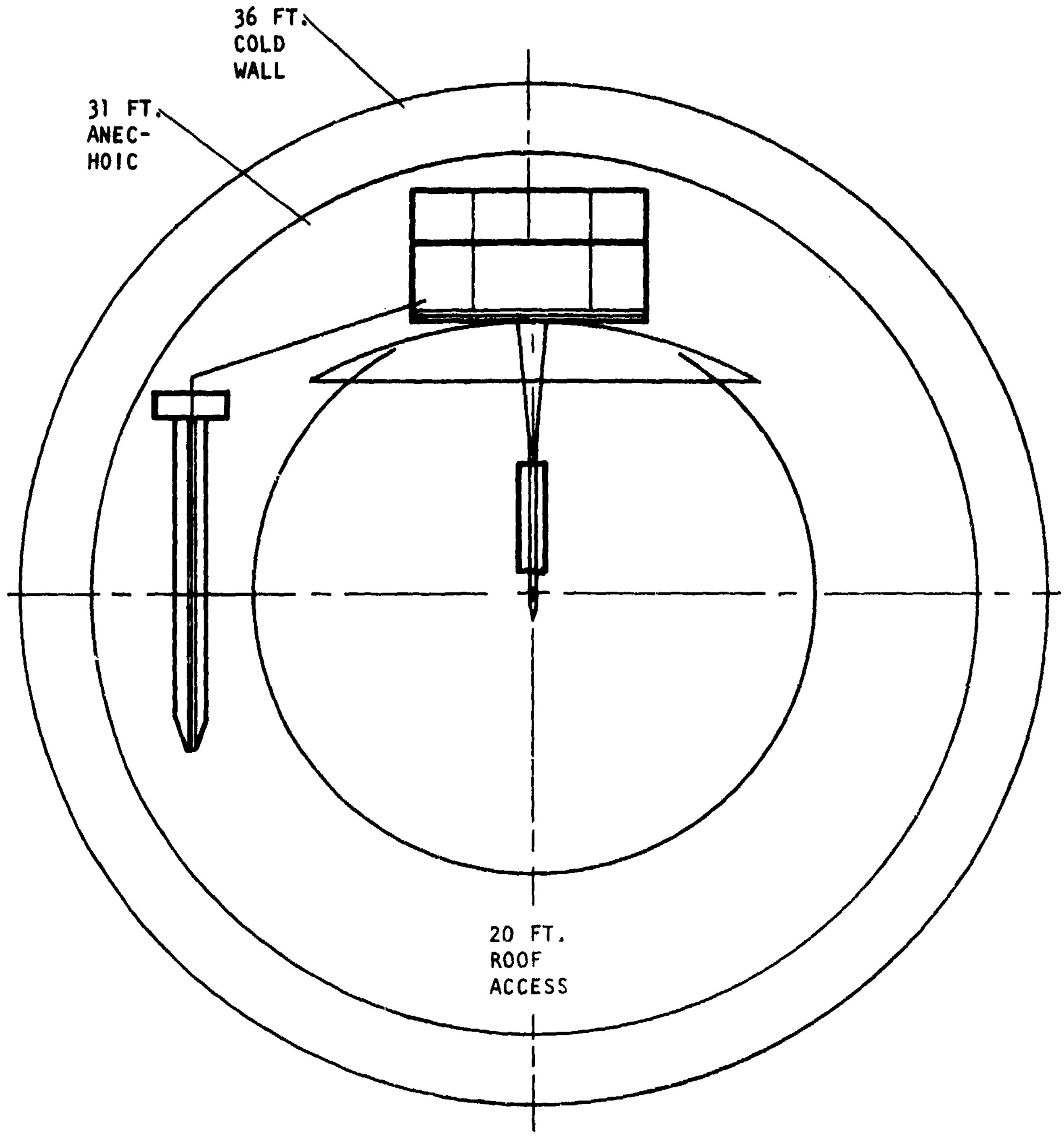



Figure A-31. Spacecraft 180° to source, center body on source axis, close to source.

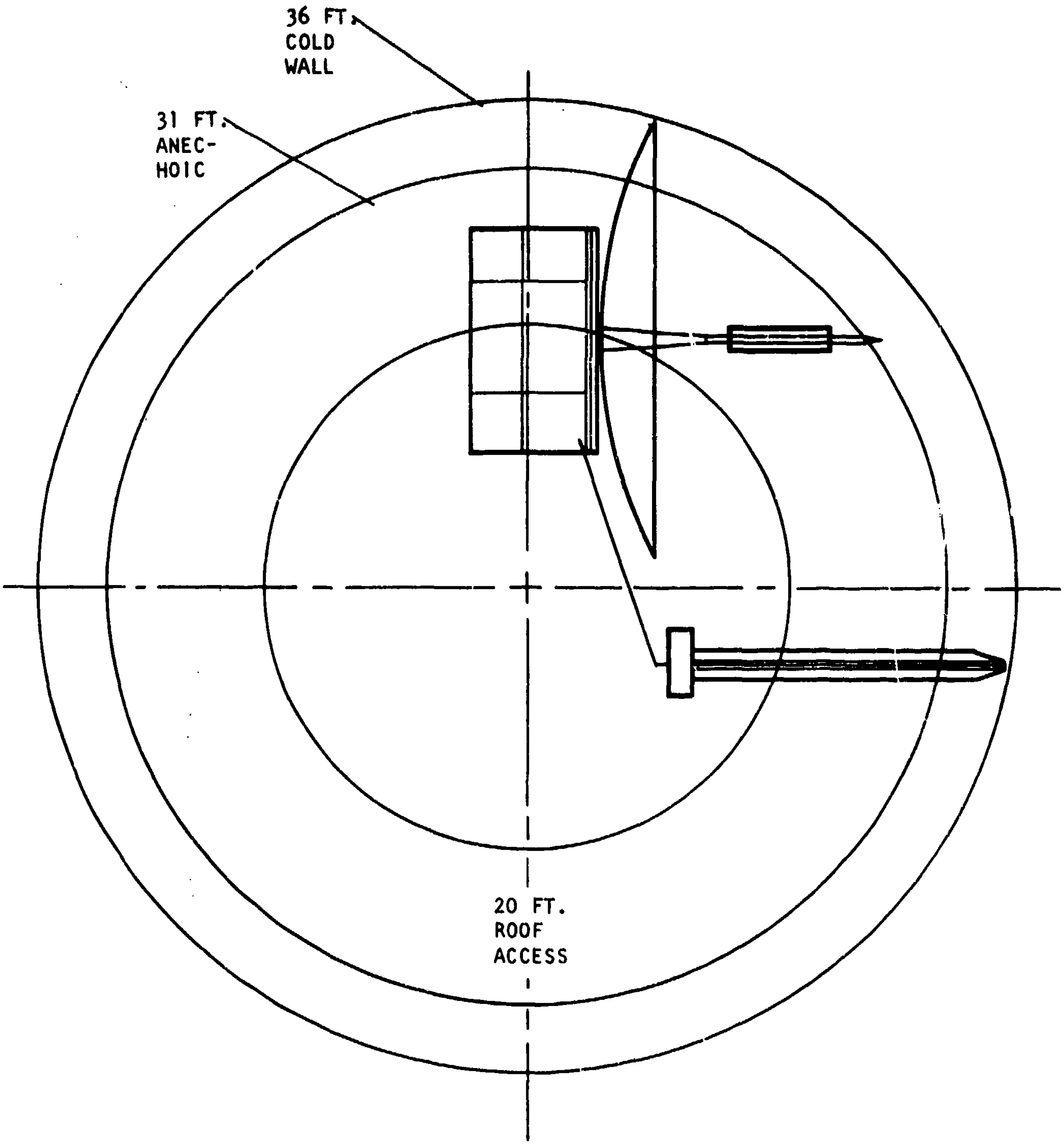



Figure A-32. Spacecraft 90° to source, center body on source axis.

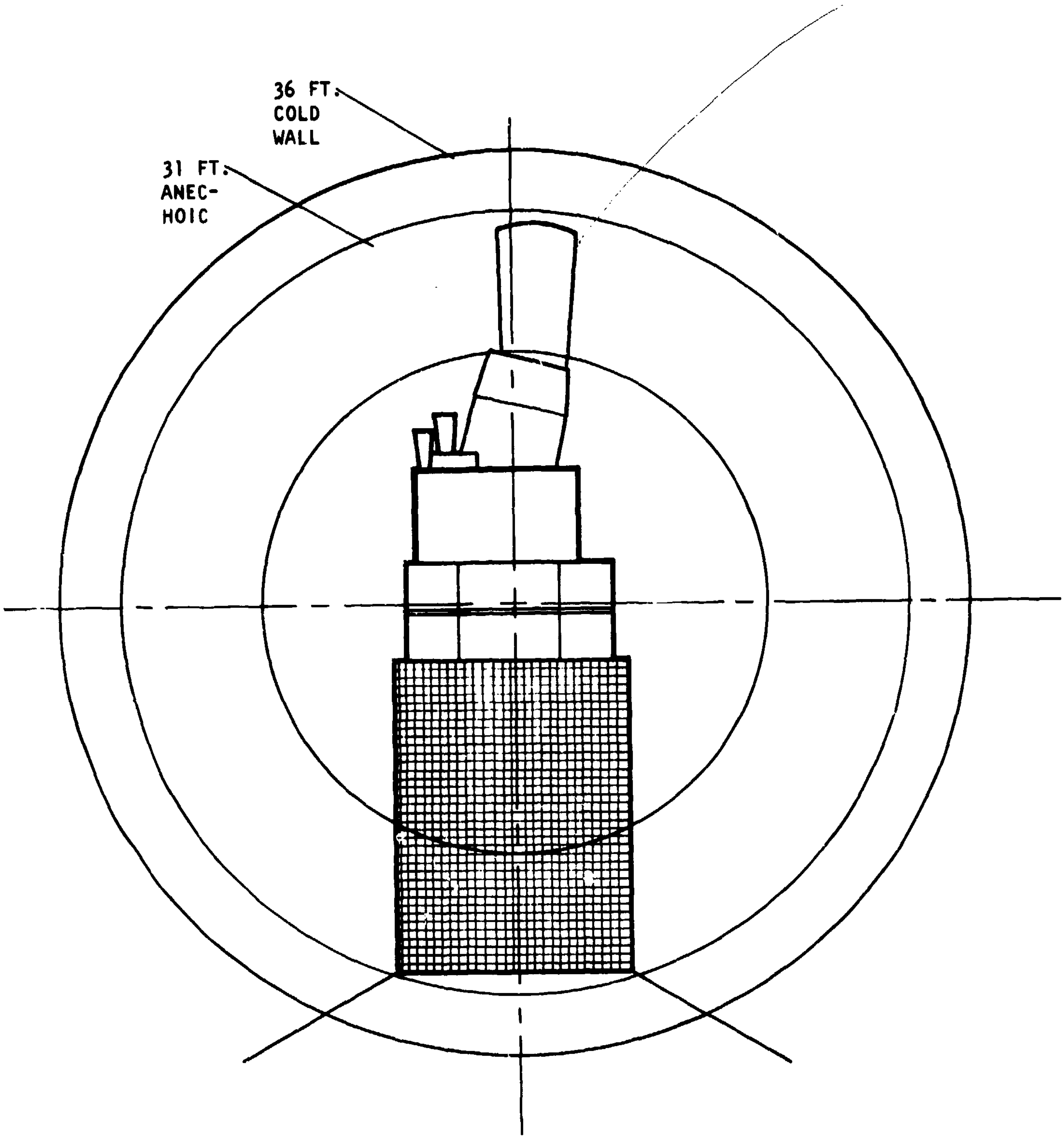



Figure A-33. Most likely exposure direction for DSP.

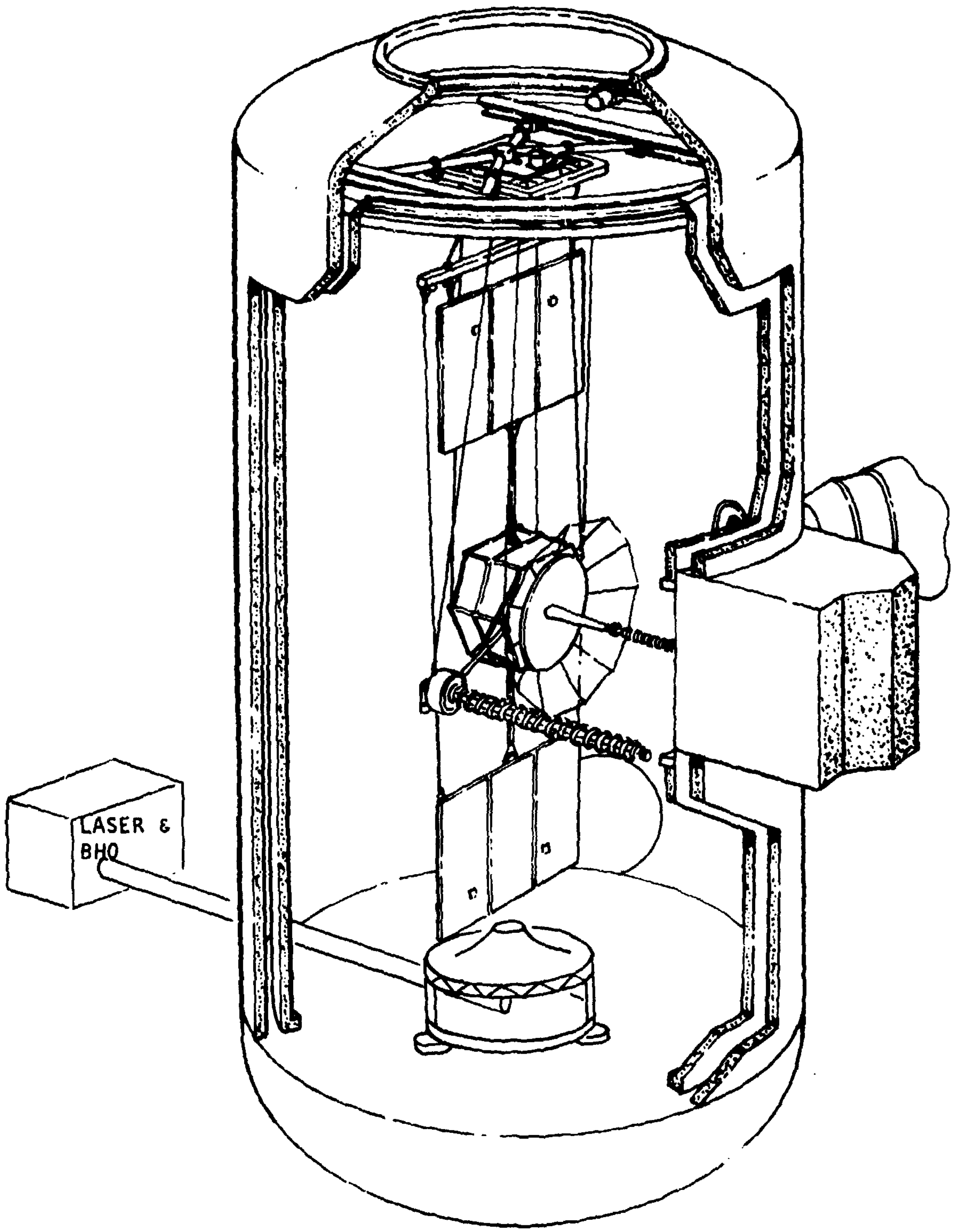



Figure A-34. Laser simulation technique for AEDC chamber.

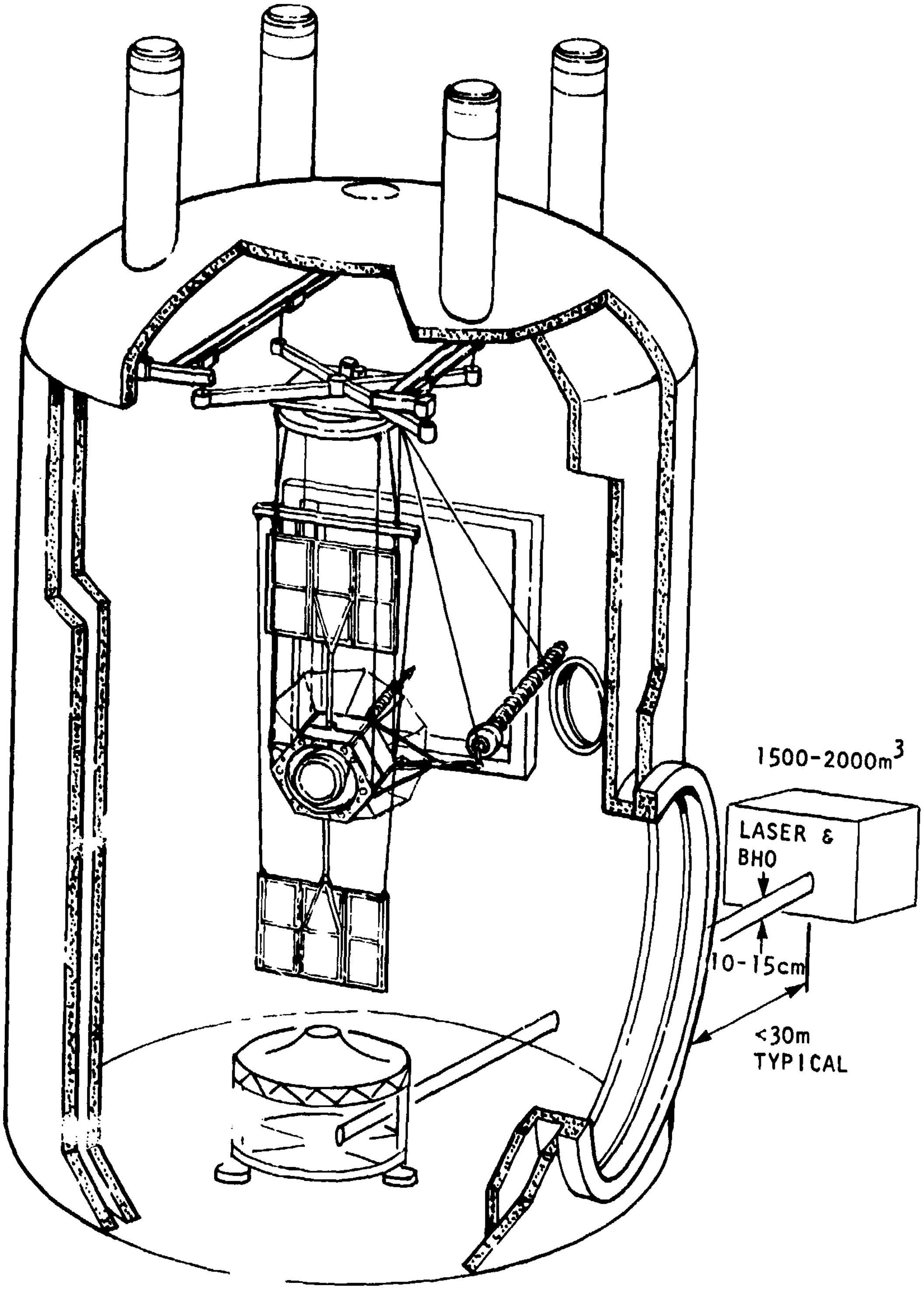



Figure A-35. Laser threat simulation technique for NASA chamber.

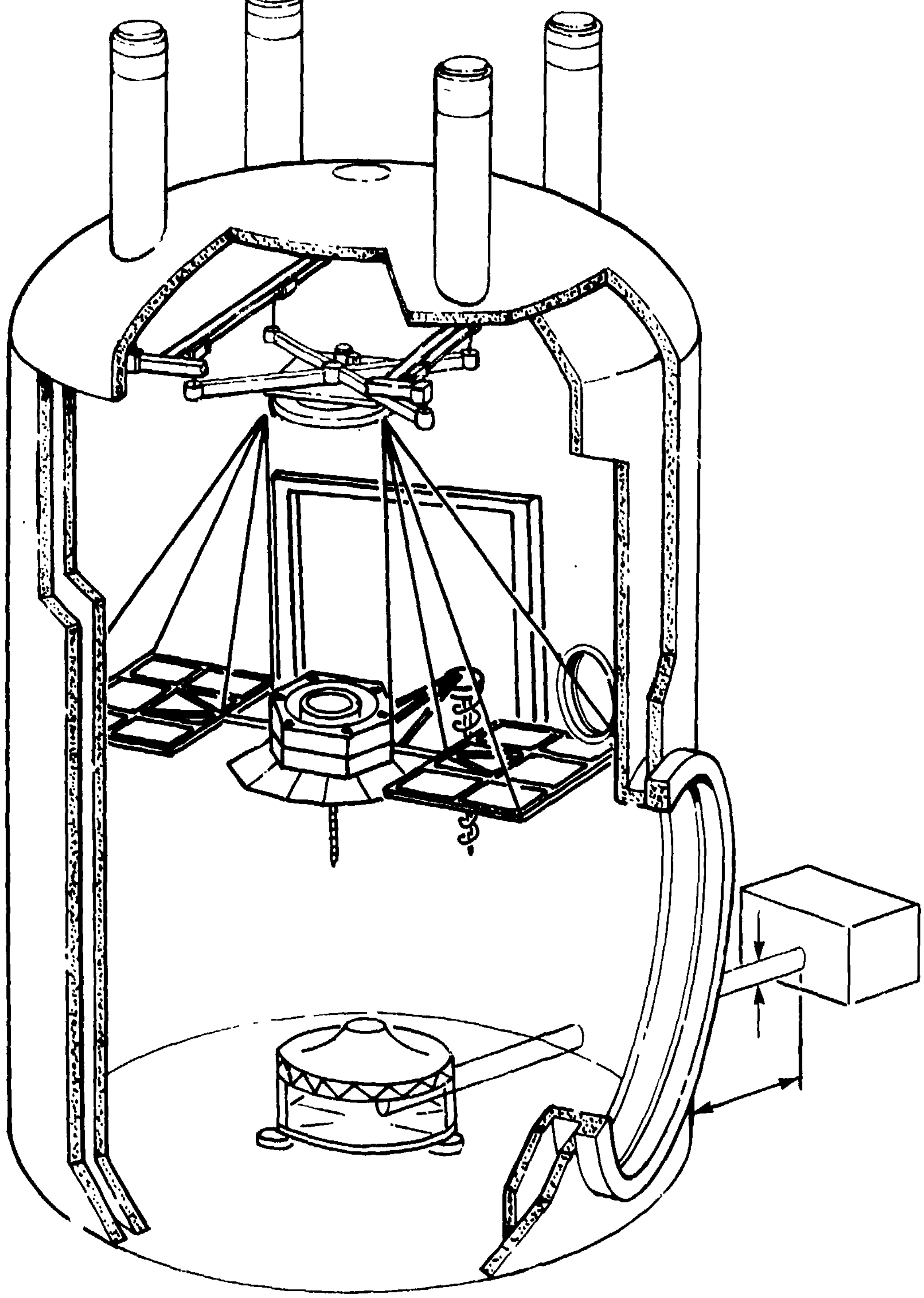


Figure A-36. Laser threat simulation technique for NASA chamber with preferred orientation of FLTSATCOM.

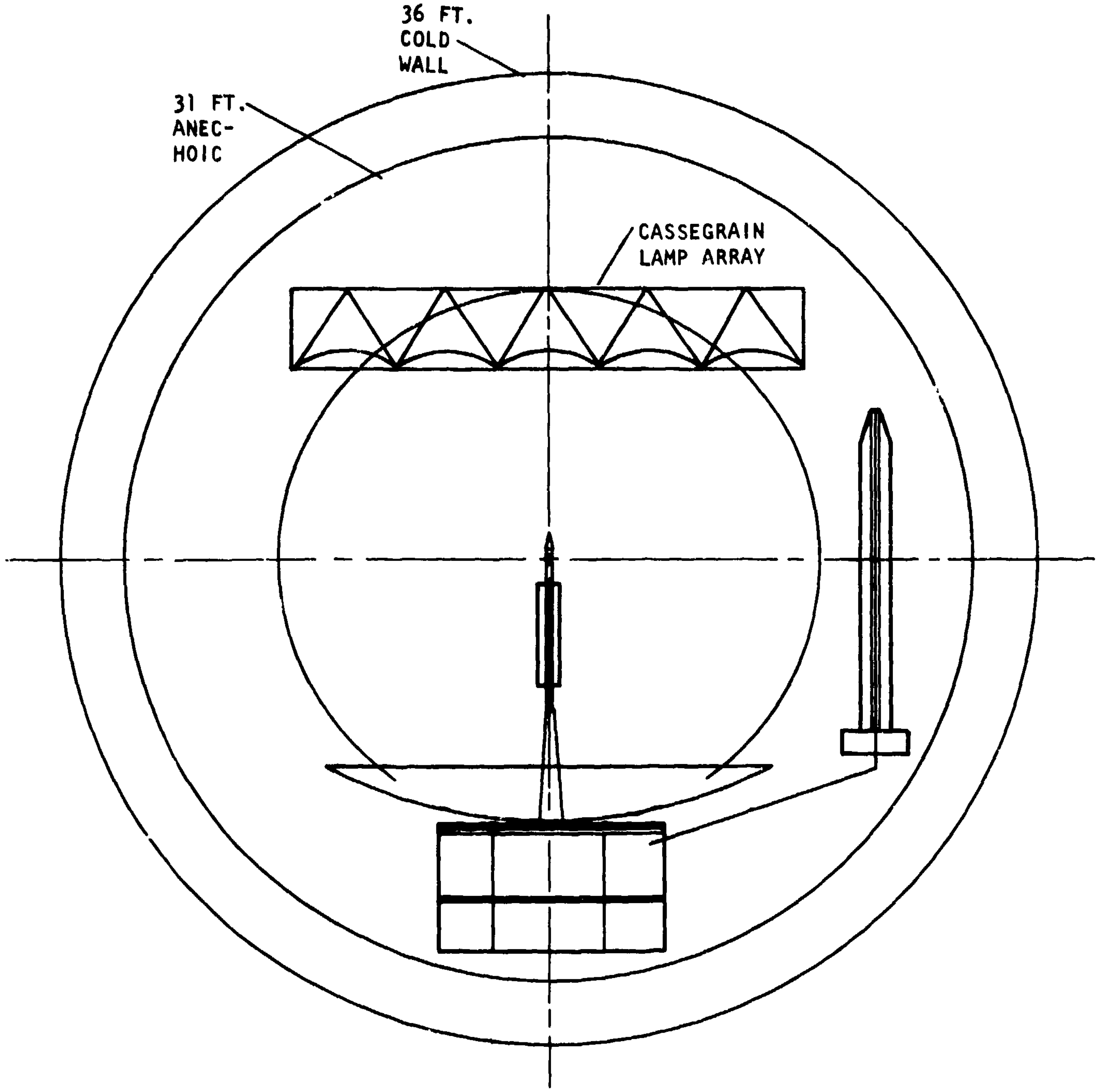



Figure A-37. Nonlaser simulated source location.

# APPENDIX B

# SXTF AUGMENTATION

# SXTF AUGMENTATION

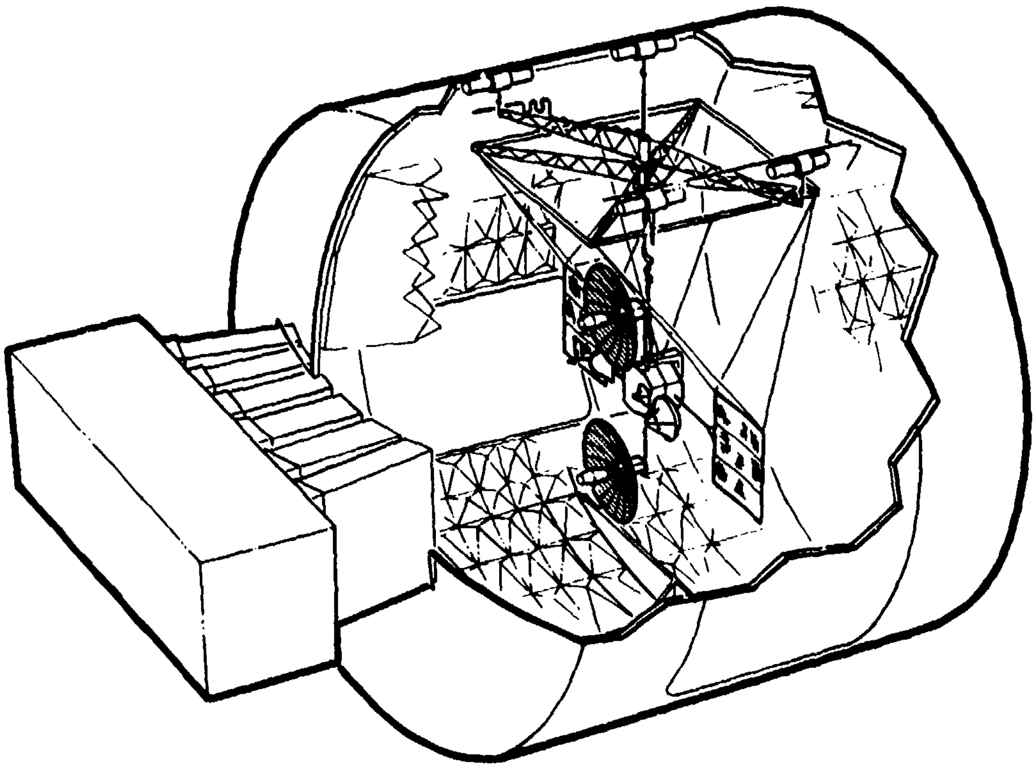


Figure B-1. SXTF augmentation.

This appendix presents the results of a study investigating alternate uses of the SXTF. The study is presented in the form of briefing slides with written commentary. (Figure B-1)

## STUDY OVERVIEW

A study was performed to support DNA in the investigation of alternate uses of SXTF. SXTF (Satellite X-ray Test Facility) as currently planned is a system level facility for the testing of satellites to X-rays. The facility consists of a large thermal vacuum chamber, X-ray sources, electron sources, satellite preparation facilities and laboratory/support areas. In this study we have introduced the concept of an integrated weapons effects test facility. The idea is to take advantage of the facility to test for all weapons effects, not just X-rays. The study is organized as follows:

First, we define the objectives and rationale for the study.

Second, we describe the additional threats considered in the study. These additional threats are lasers,pellets and electronic warfare (EW) as well as the already planned X-ray capability. Under each threat we discuss the attack modes, the effects on systems, the countermeasures and the testable uncertainties.

Third, the potential role of the operational satellite ground station in survivability is reviewed. We also discuss the role it might play in survivability testing.

Fourth, we describe some of the features of the facility that might be used for integrated weapons effects testing.(Figure B-2)

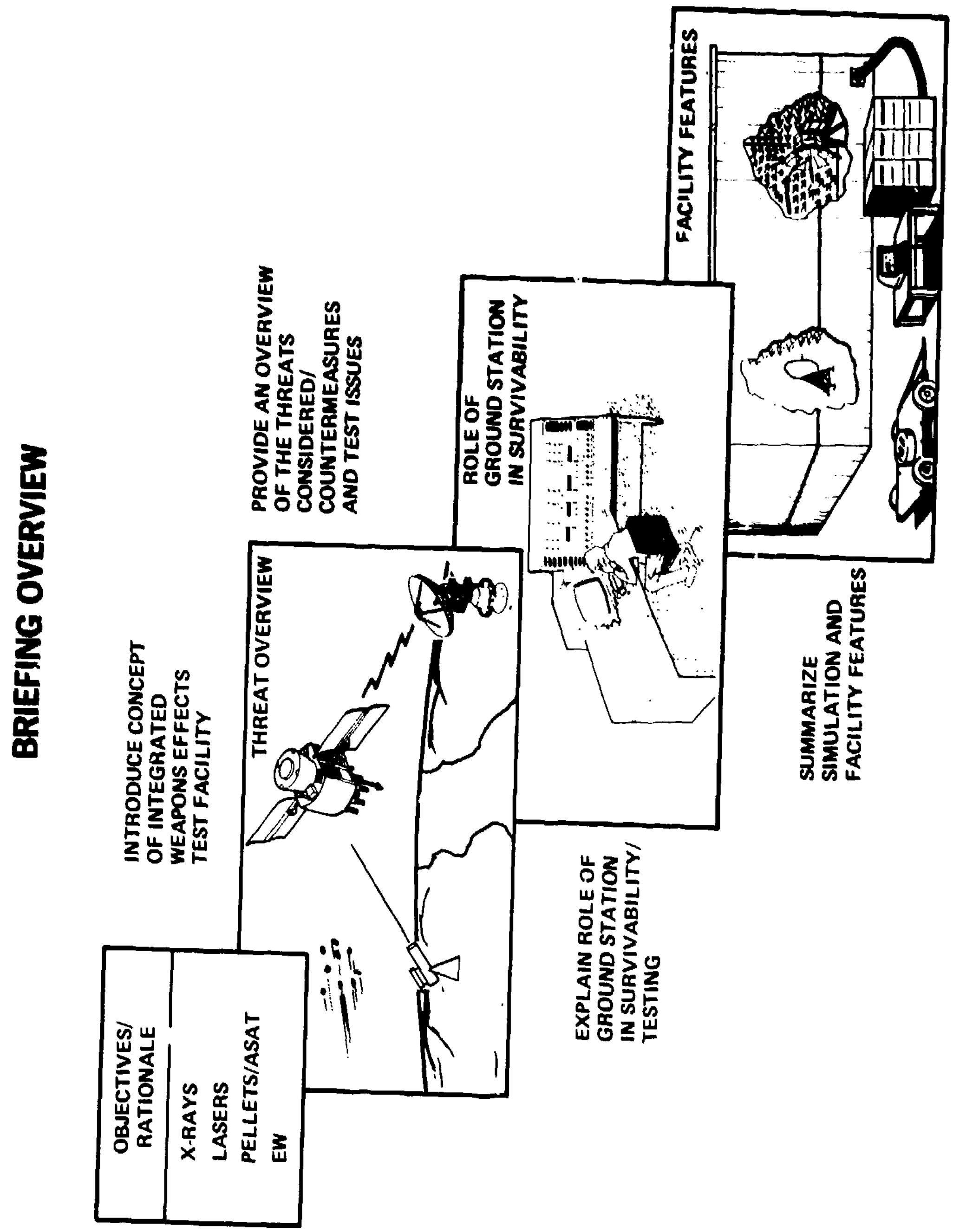



Figure B-2. Briefing overview.

## STUDY OBJECTIVES

The objective of this study is to discuss the rationale for and features of an expanded SXTF. The additional weapons effects test capability which could be added would make SXTF into an integrated weapons effects test facility. One motivation for the study of additional capability is the lack of existing test capability for satellites. A second is the significant potential cost effectiveness of combining testing facilities. We show significant technical and verification issues for each of the weapons effects considered. System testing is indicated to resolve many of these uncertainties. The cost and schedule impact of any systems testing on satellite hardware is significant. By collocating facilities the cost and schedule effects can be minimized. Given a requirement to test against several weapons effects, the collocation also helps to minimize spacecraft shipping and handling (which typically are very risky operations for delicate spacecraft hardware). Further, we will see that there are many common elements required for testing to the various threats. (Figure B-3)

## STUDY OBJECTIVES/RATIONALE

OBJECTIVE

IDENTIFY ADDITIONAL WEAPONS EFFECTS CAPABILITY FOR CONSIDERATION AT SXTF

RATIONALE

SIGNIFICANT WEAPONS EFFECTS IN ADDITION TO X-RAYS

NO SYSTEMS TEST FACILITIES FOR THESE EFFECTS

COST EFFECTIVE TO COLLOCATE FACILITIES

- USE COMMON ELEMENTS AND INTEGRATE TESTING
- MINIMIZE HANDLING/SHIPPING

Figure B-3. Study objectives/rationale.

## BACKGROUND

U.S. defense planners are concerned about the development of high energy lasers (HELs) by foreign powers and the threat that such devices pose to strategic U.S. satellite systems. The response has been a number of material technology programs aimed at developing hardened materials and construction techniques to counter the projected near term Soviet laser threat. (Figures B-4 and B-5)

The natural evolution of HEL technology has resulted in both a strengthening and broadening of the HEL threat for the period beyond 1980. The projected threat for this period includes the wavelengths of all the known HEL devices and the power necessary to deliver much higher peak irradiances to low orbits from both ground and airborne platforms. In addition, the technology required to deploy multi-megawatt lasers in space is rapidly being developed. Finally, high energy pulsed lasers create a new dimension in the threat by extending it to the visible wavelength range while retaining the wavelengths mentioned above. The pulsed threat further complicates matters because target response is markedly different in general to the much higher peak intensities and shorter exposure times associated with pulsed irradiation.

Efforts have been primarily aimed at developing countermeasures (CMs) which permit the target to survive or tolerate a direct attack. Unfortunately, there are limits to this approach. Beyond a certain threat level the incremental weight/cost penalty for a given increment in survivability becomes excessive. Hence, efforts are underway to develop CMs which make it much more difficult for a irradiation to occur. The optimum mix of Threat Tolerant (TT) and Threat Avoidance (TA) CMs for a satellite depends strongly on mission, orbit and other satellite operational constraints.

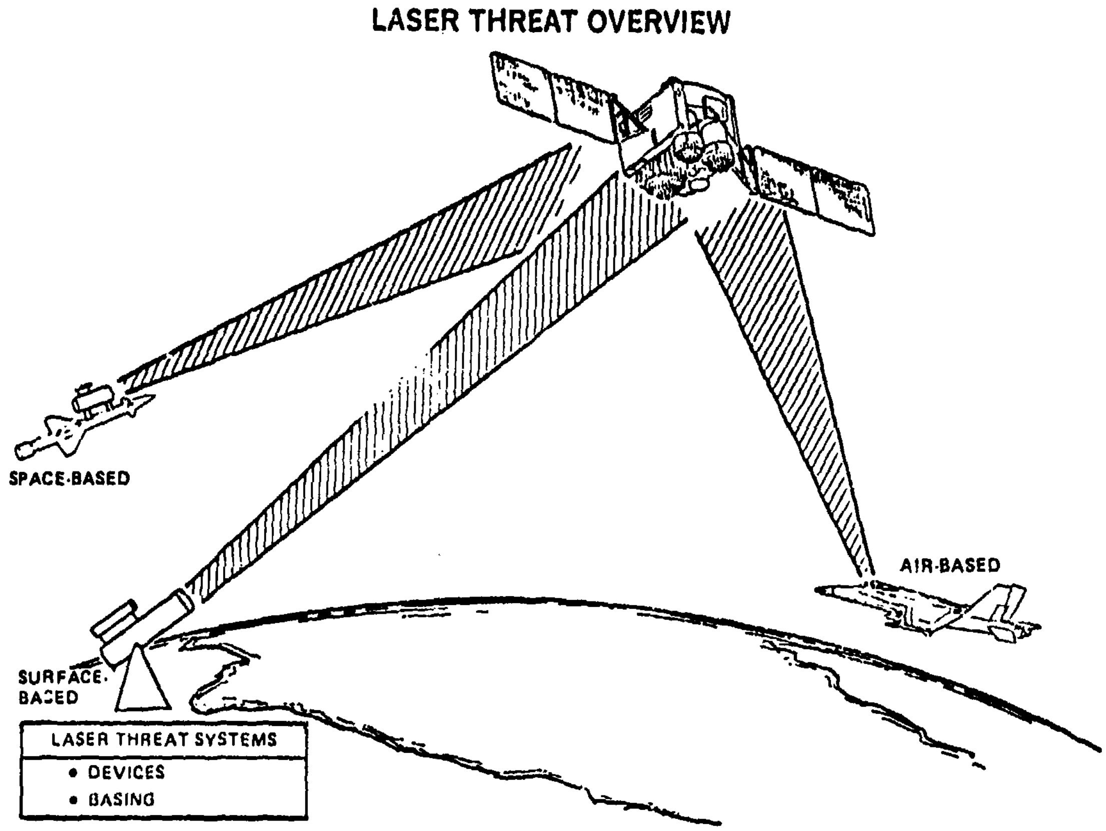



Figure B-4. Laser threat overview.

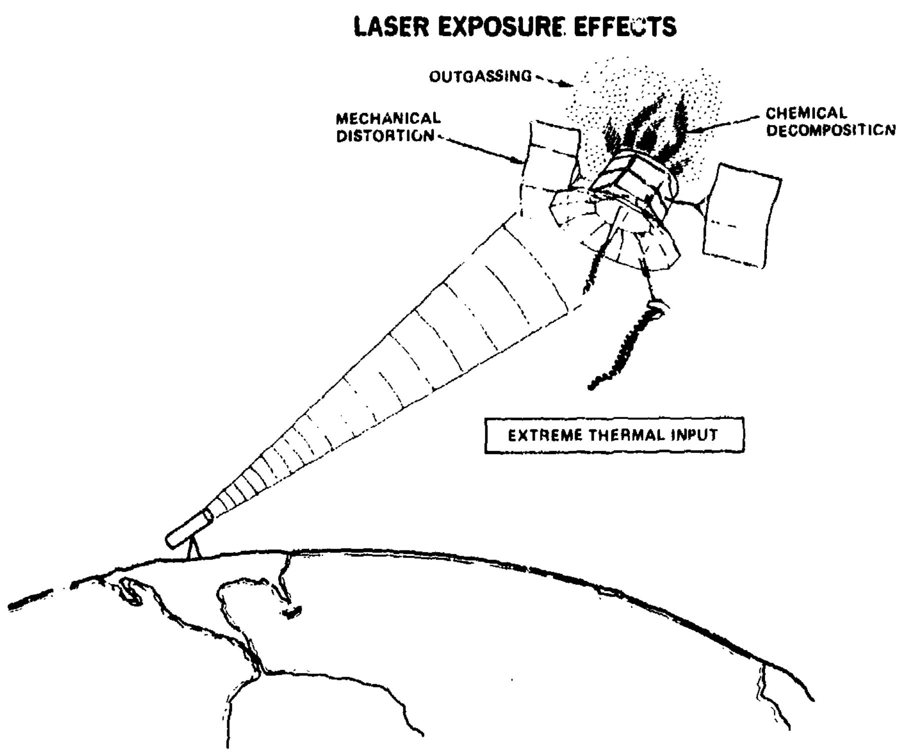



Figure B-5. Laser exposure effects.

## THREAT AVOIDANT COUNTERMEASURE APPROACHES

Several general TA CM approaches have promise. These approaches counter either the laser beam itself or the acquisition, pointing and tracking (APT) functions of the threat system.

The laser beam can be deflected by drawing it off to a preferred target location such as a shield or hard target. Or the laser beam can be intercepted/diffused by a cloud of aerosol particles or chaff.

The APT threat functions can be confused or negated by aim-point proliferation (decoys, illuminated chaff), or directly attacked through passive fire return (cube corner retroreflector) or active jamming (on-board laser). These latter approaches attempt to produce spurious targets in the APT signal processing electronics through sensory overload or on the detector plane through scattering and internal reflections inside the APT optical system. Under some conditions, the APT sensor system could itself be damaged by the unexpectedly high signal return from the target. (Figure B-6)

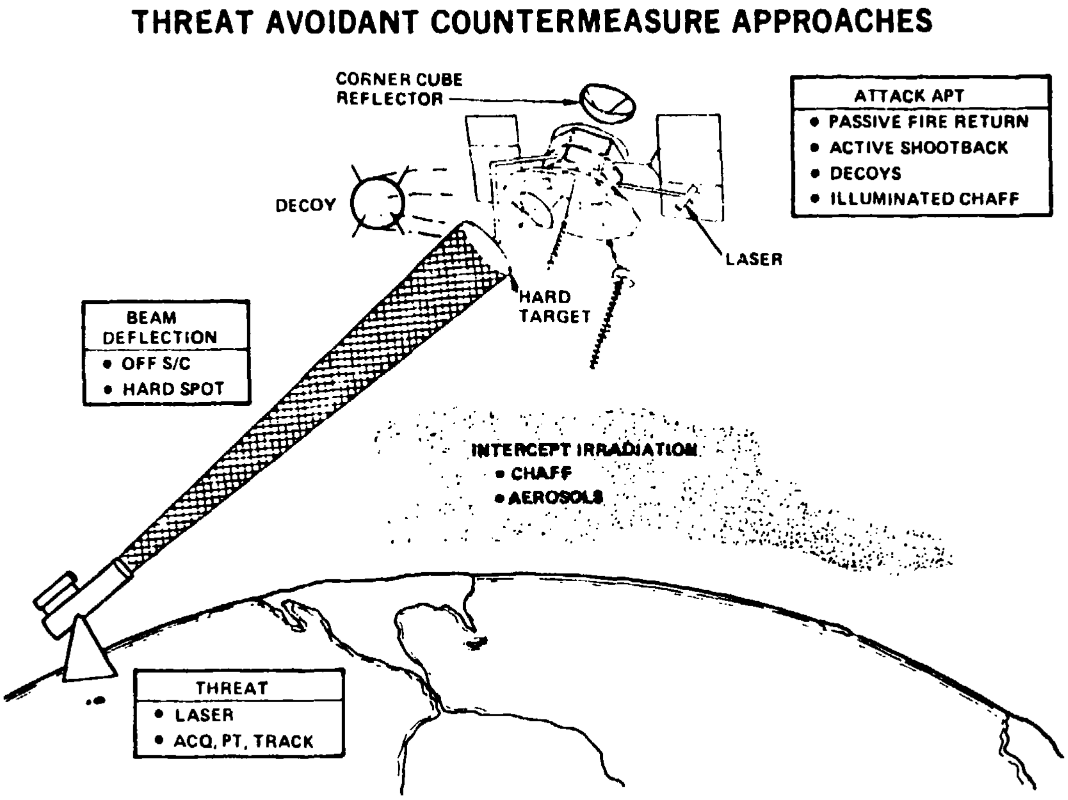



Figure B-6. Threat avoidant countermeasure approaches.

## THREAT TOLERANT COUNTERMEASURE APPROACHES

Several general TT CM approaches have shown promise and are reasonably effective. They rely generally on rejecting the incident HEL radiation or increasing the thermal mass of the exposed surface to a level at which tolerable temperature excursions are produced by the irradiation.

Simple reflective coatings and filters are not generally adequate to reject the threat. First, vulnerable surfaces must generally also absorb or transmit at some wavelengths (e.g., solar cell covers must transmit in the visible spectral region). Second, the laser threat is not at a single discrete wavelength, but rather a relatively wide band of wavelengths. Hence, specially designed and very complex multi-wavelength selective filters are required. Another approach is to permit the exposed surface temperature to increase but thermally de-couple the outside surface from the internal temperature-sensitive components through the use of multilayer blankets.

Threat tolerance can be increased most readily by merely changing to high temperature materials where possible, though only a modest increase in hardening is possible. Where the radiation must be absorbed, the addition of phase change materials to surfaces or substrates reduces temperature excursions by translating most of the absorbed energy into a change of phase. (Figure B-7)

# THREAT TOLERANT COUNTERMEASURE APPROACHES

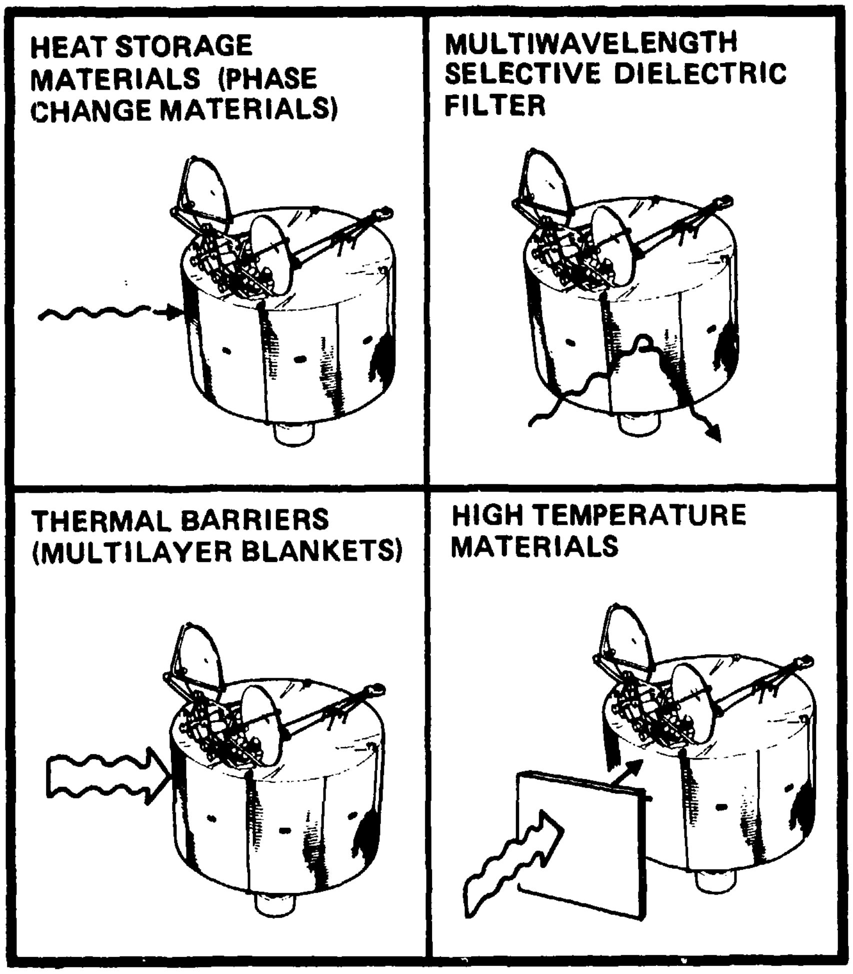



Figure B-7. Threat tolerant countermeasure approaches.

## TESTABLE UNCERTAINTIES - LASERS

The fully integrated satellite system response to laser irradiation has not been experimentally verified. All system vulnerability data have been extrapolated from small sample testing and component performance estimates. Optical (i.e., visible and infrared) cross-sections or signatures of satellites have also not been experimentally determined. Multiple reflections can contribute significantly in both areas and can only be assessed with certainty in full-scale testing.

The effectiveness of various TT and TA CMs have been demonstrated only insofar as extrapolations from coupon testing and sub-scale modeling are valid. A considerable margin of uncertainty remains in both satellite responses and TT/TA CM effectiveness.

Ground operations awareness and response to an HEL attack is a facet of the problem which until now could not even be considered. The existence and operation of the contemplated test facility will permit resolution of all of these areas of information deficiency and considerably increase confidence that our strategic satellite systems will be able to successfully survive an HEL attack. (Figure B-8)

## TESTABLE UNCERTAINTIES—LASERS

| ISSUES | CURRENT STATUS | UNCERTAINTIES |
|---|---|---|
| MATERIALS RESPONSE | 10-100 $CM^2$ AREA SAMPLES TESTED | SCALING TO SYSTEM<br>UNANTICIPATED RESPONSES |
| FULL STRUCTURE RESPONSE | NO TESTS | HEAT FLOW<br>RERELECTIONS<br>OPTICAL SIGNATURE |
| EFFECTIVE COUNTERMEASURES | NO TESTS UNDER THREAT-LIKE CONDITIONS | EFFECTIVENESS OF<br>– THREAT TOLERANCE<br>– THREAT AVOIDANCE |
| GROUND AWARENESS | NO TESTS UNDER THREAT-LIKE CONDITIONS | ABILITY TO<br>IDENTIFY ATTACK<br>INITIATE COUNTER-MEASURE<br>ASSESS DAMAGE |

Figure B-8. Testable uncertainties--lasers.

## PELLET THREAT

Pellet ASAT represents a near term realizable attack mode for satellites. Three versions of the threat are considered for a pellet ASAT. First, pellets can be introduced nonexplosively into the satellite orbit. Second, an explosively activated pellet generator can be fired directly at the satellite. This might either be a "gun" as shown in the figure, or an exploding vehicle where the vehicle fragments are the pellets. The third form of the pellet threat is really not a pellet but rather, direct vehicle impact. (Figure B-9 )

There are several possible modes of attack sensing. The first would be on-board sensing either optically or by radar. The weapon might be ground based sensing and attack control, also, either optically or with radar. Third, the weapon might use the target operational signals to home in on the target.

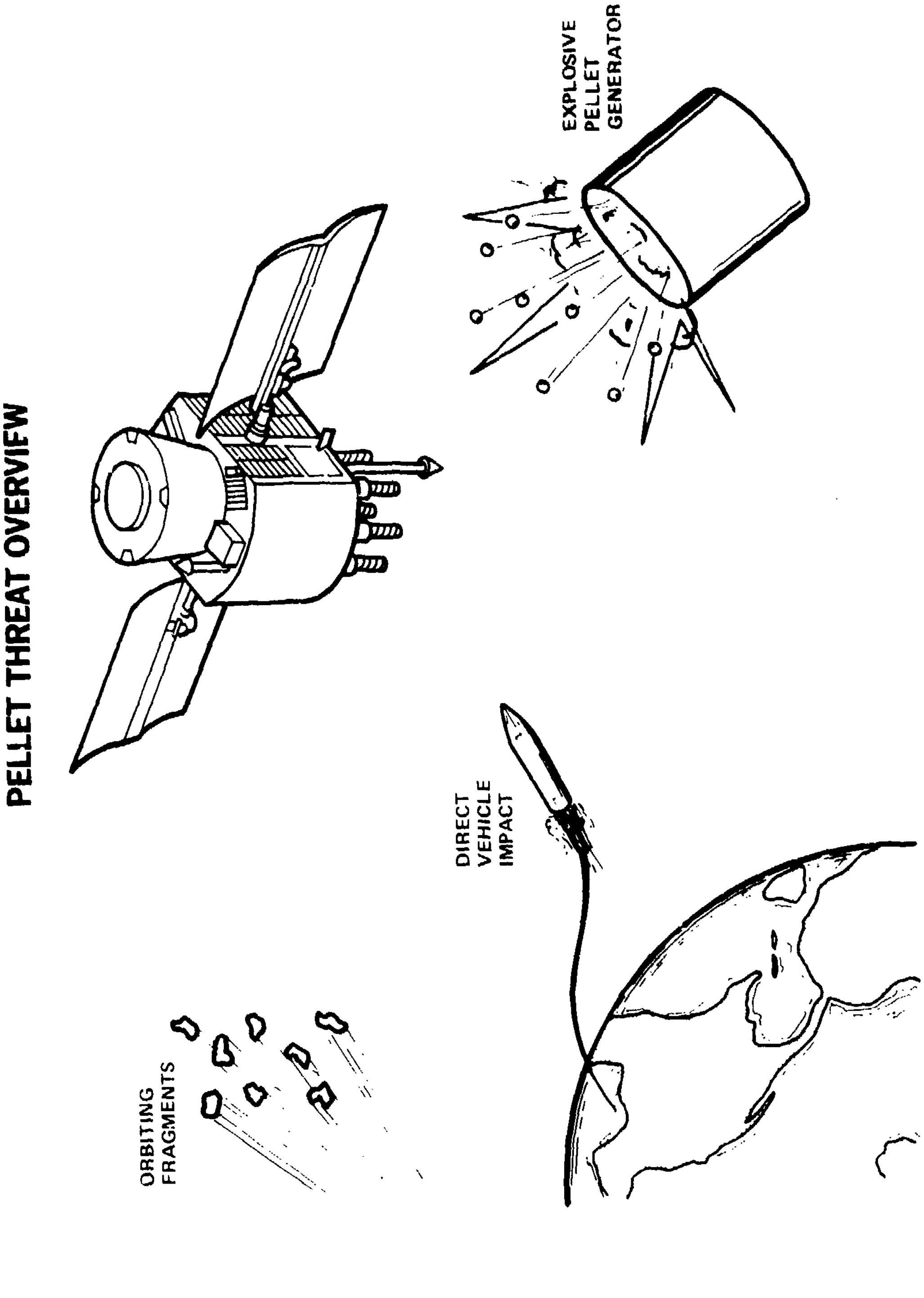



Figure B-9. Pellet threat overview.

## PELLET COUNTERMEASURES

Countermeasures follow the threat avoidant and threat tolerant breakdown. The threat avoidant techniques include:

1. ECCM - the homing function of the weapcn may be electromagnetic or optical. An on-board spoofing function may be used to confuse the homing radar/laser.
2. Expendables - confuse the homing function through the use of chaff, decoys or aerosols.
3. Move - with sufficient on-board propellant a target satellite might be able to retreat.
4. Counterattack - fire laser or recoilless rifle back at attacking ASAT.

The threat tolerant techniques include the use of hard targets or preferred satellite sides. (Figure B-10)

Both the threat avoidant and threat tolerant schemes rely heavily on attack sensing. The sensing of attack then becomes a very key feature in the survivability. The sensing might be based on thermal sensors for blast or electromagnetic/optical sensors for the homing device.

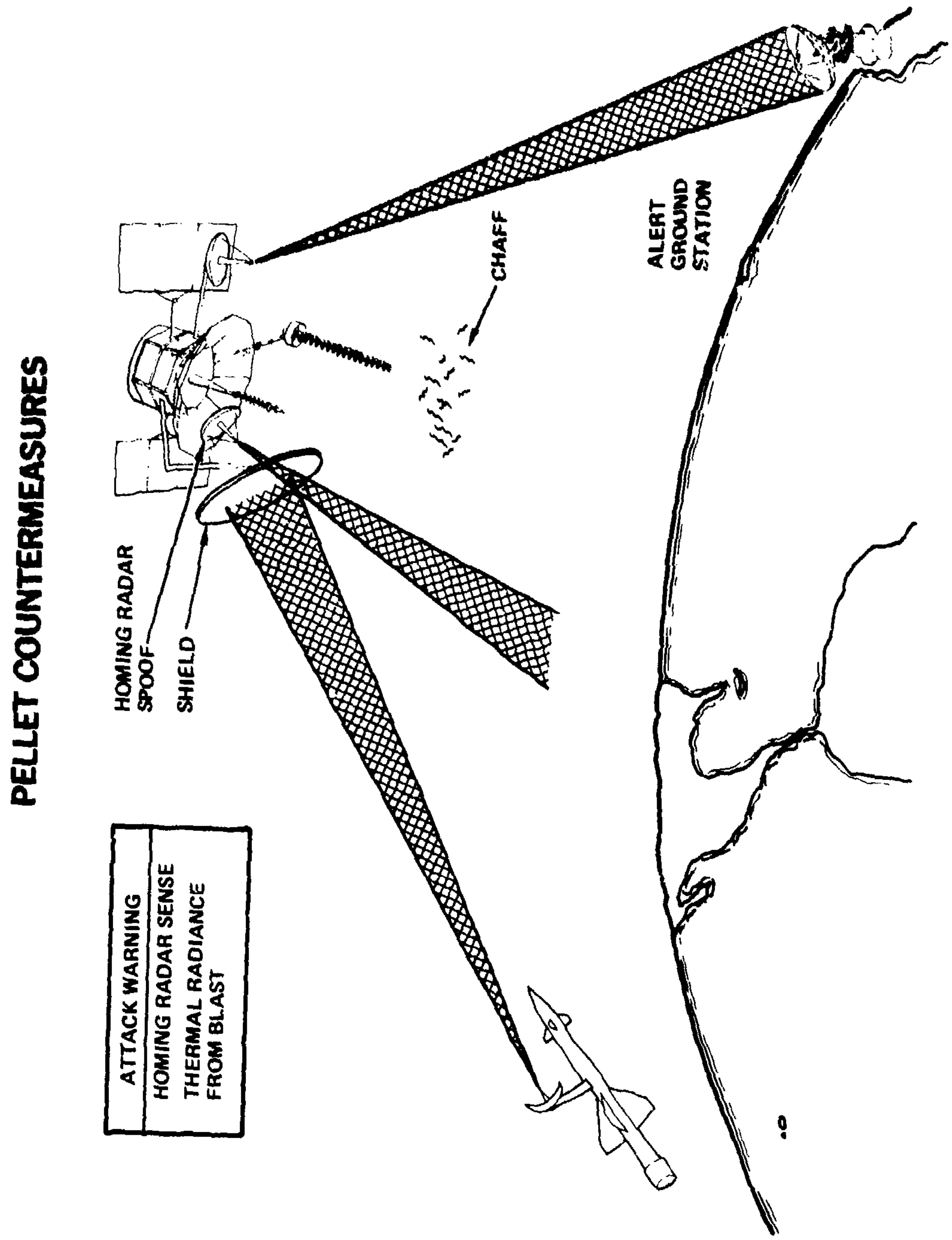



Figure B-10. Pellet countermeasures.

## UNCERTAINTIES - PELLETS

Little real development work has been accomplished in pellet countermeasures. Therefore, the uncertainties are broad and not very specific. First, many potential countermeasures depend on the ability to sense the presence of the threat, either through electromagnetic or optical means. Since these measures are largely untested, they are certainly candidates for evaluation. Simulations of the attacking vehicle parameters would be required in the presence of the spacecraft, and telemetry would be monitored to see whether appropriate action was taken.

A second uncertainty has to do with the ability of a spacecraft to assess its own damage or the ability of the ground station to infer damage from the available signals. This is a realistic concern since a pellet attack might not necessarily be lethal. An example might be the case of such a low density attack that only a few pellets struck the spacecraft. With the extensive redundancy on board, damage to one or two electronics boxes, or to a few strings of solar cells would not mean mission failure. Testing could determine whether a hole through a box is lethal or whether structural damage is recoverable.

Another verification issue would be the effectiveness of the countermeasures. These would untimately become part of the normal operating functional parameters of the spacecraft. However, the checkout of these functions may require special conditions such as an anechoic chamber for ECCM. (Figure B-11)

## UNCERTAINTIES—PELLETS

| ISSUES | CURRENT STATUS | UNCERTAINTIES |
|---|---|---|
| ATTACK SENSING<br>THERMAL RADIATION<br>EXPLOSIVE PRODUCTS<br>PELLETS<br>HOMMING RADAR | NO TESTS | ABILITY TO WARN OF ATTACK |
| DAMAGE ASSESSMENT | NO TESTS | GROUND STATION KNOWLEDGE OF SATELLITE CONDITION |
| EXTENT OF DAMAGE | NO TESTS | DOES ENCOUNTER IMPLY LOSS OF FUNCTION |
| EFFECTIVENESS OF COUNTERMEASURES | NO TESTS | DO COUNTERMEASURES WORK AS DESIGNED ON TIME |

Figure B-11. Uncertainties--pellets.

## ELECTRONIC COUNTERMEASURES OVERVIEW

Satellites use electromagnetic links for most of the interactions with the user. These include:

Command/programming up and downlinks
Communications up and downlinks
EM and Optical sensor data downlinks
Housekeeping telemetry downlinks

Because these links are electromagnetic they are potentially susceptible to electronic warfare. The threats, in increasing order of sophistication, are:

| | |
|---|---|
| Transmission Intercept (SIGINT) | - Adversary just listens |
| Link Jamming | - Adversary generates interference |
| Link Deception | - Interference to confuse |
| Link Exploitation | - Adversary uses the link/satellite |
| Satellite Mitigation | - Generate commands to render satellite temporarily incapacitated |
| Satellite Takeover | - Active command takeover of satellite |
| Satellite Damage | - Generate sufficient power to damage receivers |

The basing for ECM can be land, sea, airborne or satellite. (Figure B-12)

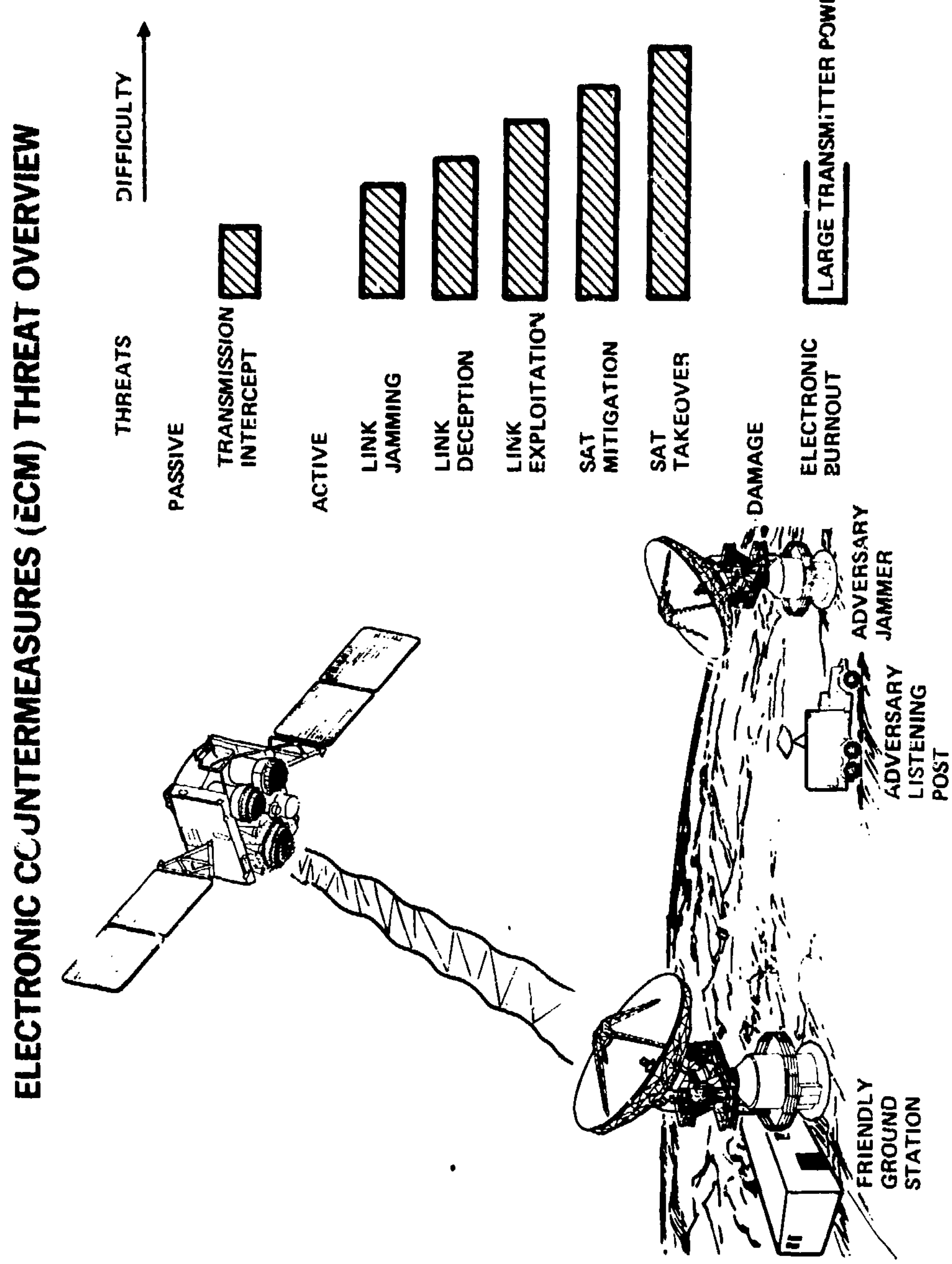



Figure B-12. Electronic countermeasures (ECM) threat overview.

## PASSIVE LINK INTERFERENCE

DEFINITION - Adversary facilities exercising their capabilities to detect, demodulate, and/or analyze signals emanating from a satellite or its associated ground support facilities. (Figure B-13)

PREREQUISITES -

- Satellite (or Ground Support Facility) visibility at times of link activity
- A priori intelligence or detection of the link signal and its external characteristics permit demodulation of the signal
- Antenna and receiver equipment with sufficient link Signal-to-Noise Ratio (SNR) to permit demodulation of the signal
- Sufficient analysis manpower to permit definition of signal internals (COMINT)

THE CONSEQUENCES -

- Passive exploitation of link data (COMINT)
- Support of active EW techniques

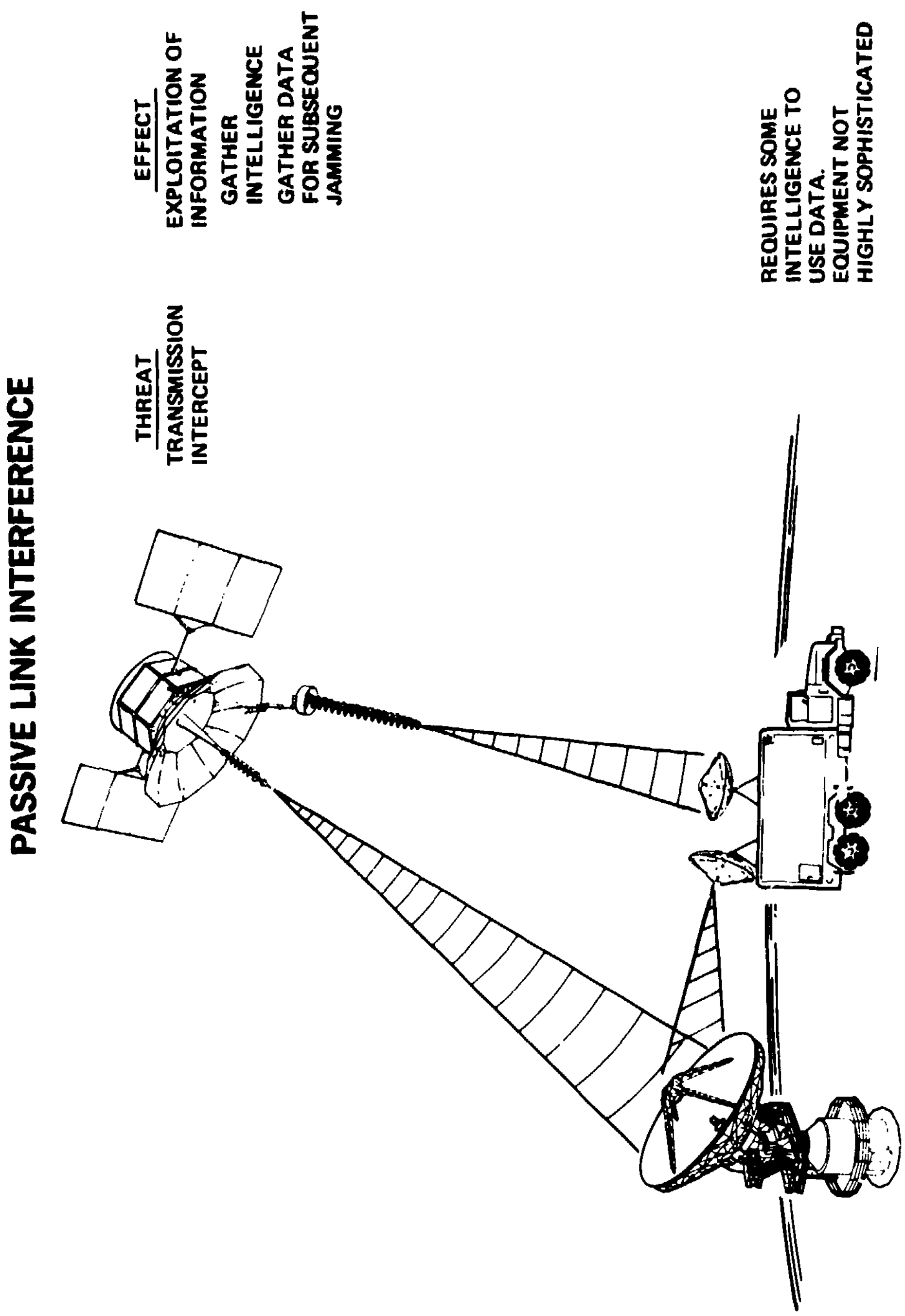



Figure B-13. Passive link interference.

## ACTIVE LINK INTERFERENCE

Active link interference can be subdivided into four principle threat areas: jamming, deception, exploitation and mitigation/takeover. Each of the threats, in order, requires increasing sophistication and preknowledge of S/C characteristics.(Figure B-14)

Jamming is the generation of interfering signals designed to harass, interrupt or negate the reception of signals normally associated with the target EM link. The prerequisites for jamming are: satellite (or intended recipient) visibility with the targeted link active, SIGINT support to assess the effectiveness of anti-link jamming against the targeted system, sufficient jammer to signal (J/S) power ratio within the targeted receiver to mask or degrade the intended signal, and sufficient knowledge of the link characteristics to overcome jam-resistant modulation/demodulation schemes. The consequences of link jamming are a degraded link performance resulting in reduced or interrupted link/sensor capabilities and the loss of satellite control.

Deception is covert link interference designed to confuse the system with false information. The prerequisites for deception are: satellite visibility during times of intended link deception with the link active, SIGINT/COMINT support to assess the effectiveness of the link deception, sufficient technical and operational knowledge of the system to generate high quality spoofing signals/EM signatures, and capability to deny satellite sensors their targeted EM emitters. The consequences of link deception are four-fold: misleading operational communications causing confused user response, denial of targeted EM emitters, generation of false EM emitter signatures, and loss of satellite control through erroneous command system response.

Exploitation is the intelligent use of a satellite's resources to the threats advantage. There are three prerequisites for exploitation: satellite visibility with link active during times of intended exploitation, COMINT/COMMS facility support to the exploitation effort, and sophisticated knowledge of the satellite EM link characteristics in order to employ effective covert/overt exploitation techniques. The consequences of link exploitation are: overt/covert exploitation of satellite communication capabilities, utilization of satellite sensor data to determine users friendly force disposition and/or operational vulnerability of adversary force disposition, and misdirection of satellite assets through command system exploitation.

Mitigation/Takeover - These threats render the satellite incapable of performing its designed function by means other than continuous jamming or physical damage. In the case of satellite takeover the threat denies use of the satellite by the intended host system and uses the asset to its advantage. For either mitigation or takeover there are three principle prerequisites: satellite visibility during periods when its command receivers are enabled, COMINT/Telemetry support facilities or ground control facilities equal to those of the host system, and sufficient technical and operational knowledge of the satellite to generate high quality commands. Additionally, takeover requires greater operational system knowledge. The consequences of mitigation are: loss of satellite attitude stability, reprogramming (or deprogramming) of command memory, denial of satellite command system access, irrecoverable loss of satellite or subsystem functional control, and disabling of system protective devices to increase other threat damage. In the case of takeover the consequences are: loss of a tactical/strategic satellite asset, employment of that asset against the host, and loss of the ability to deactivate that asset.

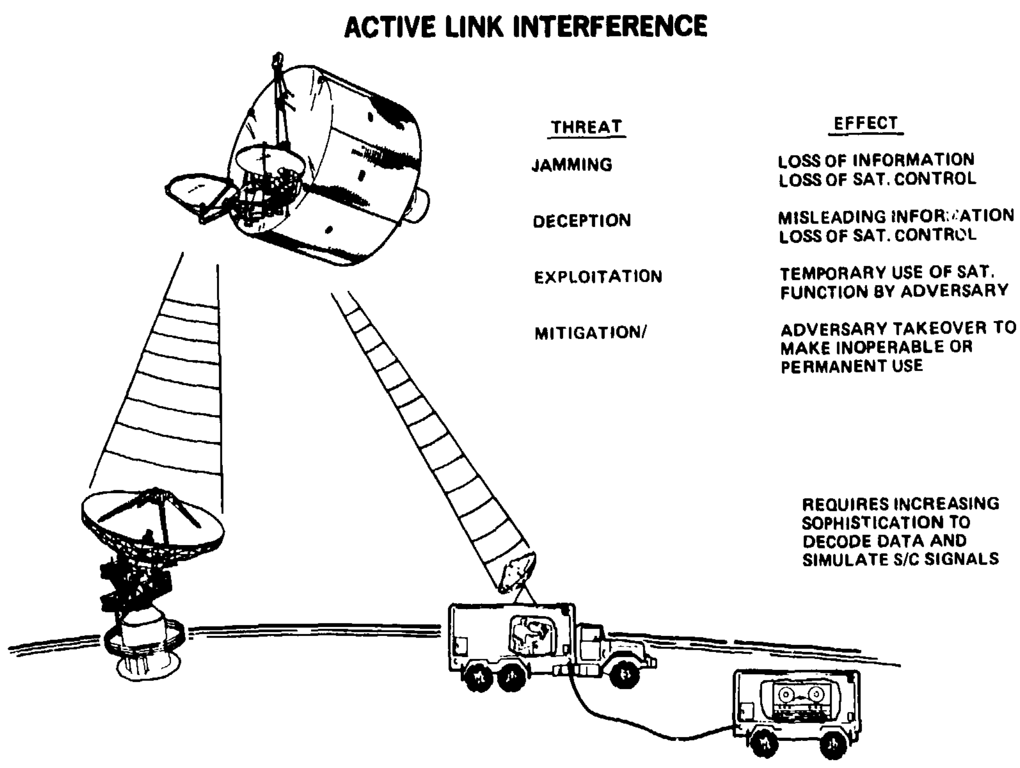



Figure B-14. Active link interference.

## SATELLITE DAMAGE

| | |
|---|---|
| DEFINITION | - Satellite EM sensor or communications subsystem degradation (or burnout) through the reception of a transient high-intensity signal overload. (Figure B-15) |
| PREREQUISITES | - • Satellite subsystem susceptibility to signal overload<br>• Technology and the facilities to generate the required transient signal intensity<br>• Sufficient technical description of the targeted subsystem configuration to determine the required signal characteristics |
| THE CONSEQUENCES | - • Degradation or loss of function of the targeted satellite subsystem |

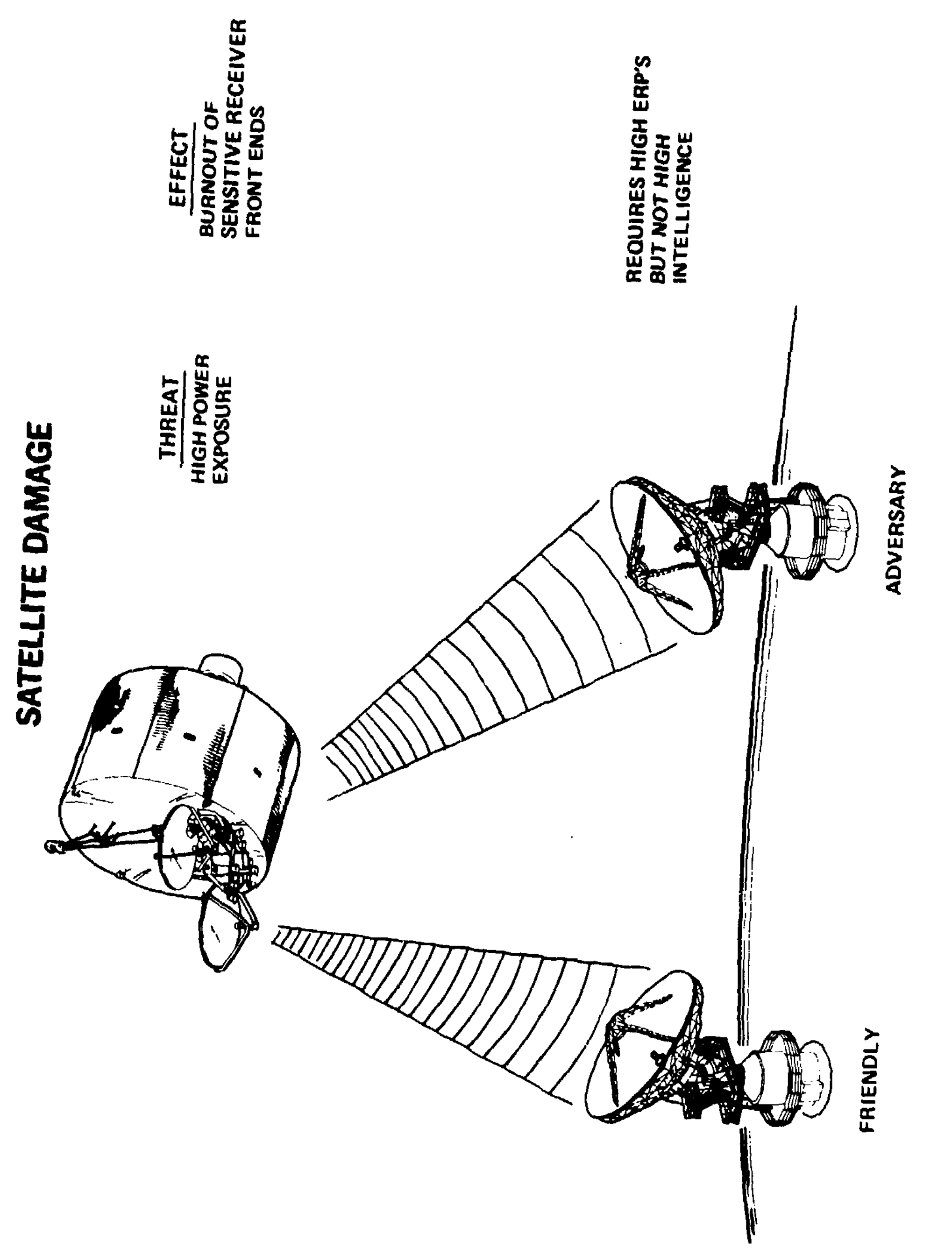



Figure B-15. Satellite damage.

## EW COUNTERMEASURES

Sidelobe suppression and spot beams reduce the signal available to adversary listening posts. Highly directive antennas are used to reduce the spill over for certain types of communications links. However, other missions prevent the use of a small earth footprint. The amount of signal transmitted in unintended directions may depend not only on the antenna directivity, but also on scatter off the spacecraft structure. (Figure B-16)

High effective radiated power (ERP) makes jamming more difficult because the jamming signal must be higher than the operational signal. The higher ERP is accomplished through higher power transmitters.

Encryption/Signal Processing makes the data being transmitted more unrecognizable to the adversary and prevents both intelligence gathering and active link deception. Signal processing rejects unwanted signals from the data.

Spread Spectrum techniques include: frequency hopping, time sequencing, and chirp. These make jamming difficult because the code is secure and the jamming power would have to be spread over such a wide spectrum that the mechanization is impractical. The low signal level also makes the signal much harder to detect.

Wartime frequencies reserves certain frequencies only for emergency use. The transmission interception and jamming then takes time to get set up after switchover.

Protective devices are used to prevent permanent damage due to very high jammer power. Clamps on receiver inputs are used to shunt damaging energy away from sensitive receiver front ends.

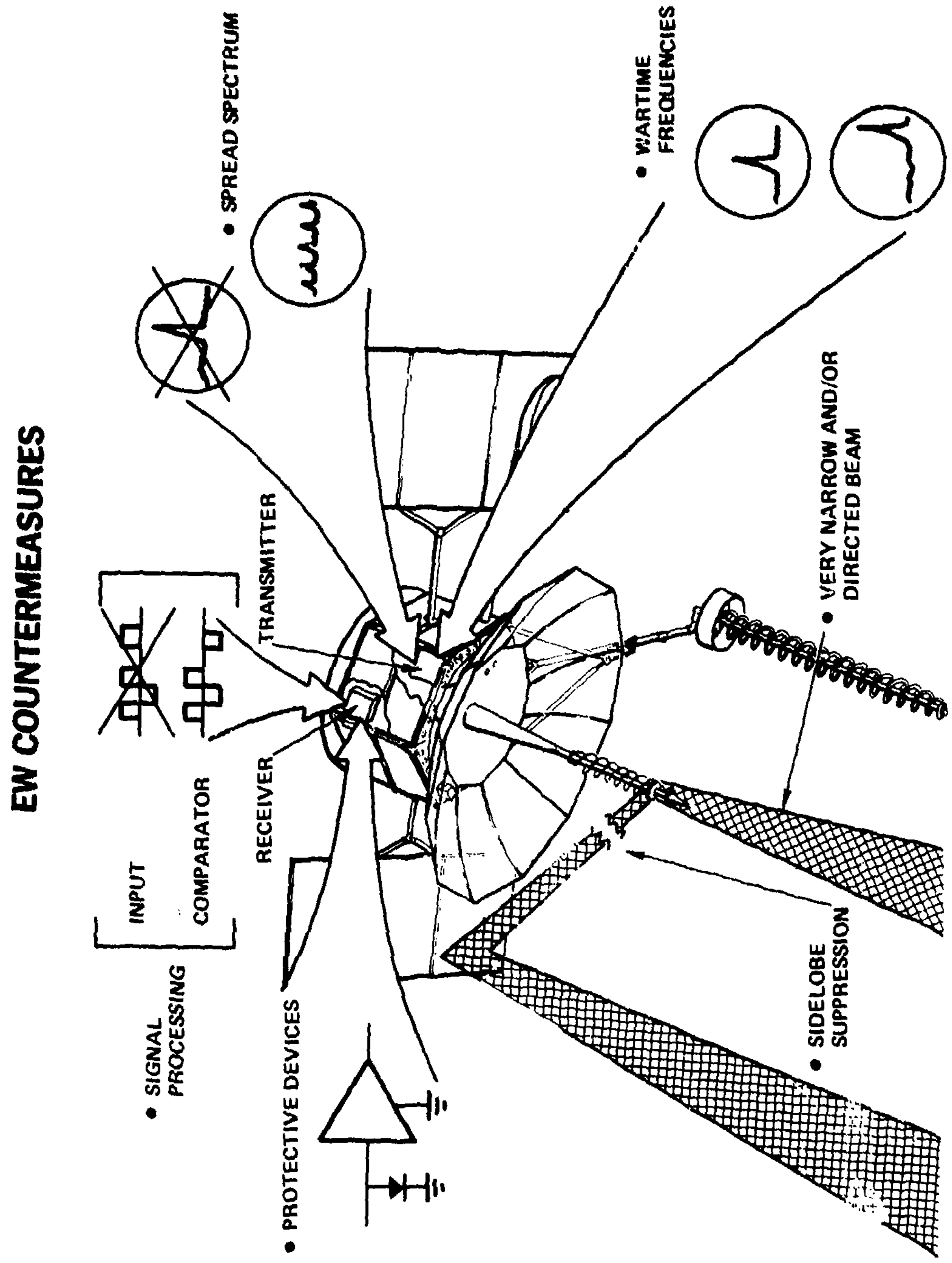



Figure B-16. EW countermeasures.

## TEST UNCERTAINTIES

The effectiveness of the ECCM would require testing both in the engineering development of the measures and in the verification phase. Some TEMPEST testing is currently performed on boxes to determine whether codes are secure. Also, the performance of specific signal processing and spread spectrum design features are part of the normal evaluation of the satellite system.

Testing is not typically performed on entire systems. One system test mode of significance would be total system radiation patterns (e.g., with solar arrays attached). The radiation pattern is affected by not only the antenna but also other pieces of the structure such as solar arrays. These appendages could spoil the very narrow beam produced by the antenna.

Total system resistance to probes and rejection of invalid signals are a concern. Some of the active features of ECCM may require accurate representation of the ground station. The satellite to ground station link could be tested with a variety of intelligent and high power jammer signals to evaluate performance.

Another issue is the recovery of protected receivers from high energy damage attempts. Overload protection would not prevent the system from going into saturation. Recovery times and multiple exposure effects would then become important. (Figure B-17)

## TEST UNCERTAINTIES—ECM

| THREAT | ELECTRONIC COUNTER-COUNTERMEASURES (ECCM) | HARDWARE TESTABLE UNCERTAINTIES |
|---|---|---|
| LINK EXPLOITATION | ENCRYPTION | TEMPEST |
| | SPOT BEAM/SIDE LOBE SUPPRESSION | EFFECTIVENESS OF SIDELOBE SUPPRESSION |
| LINK JAMMING AND SAT. MITIGATION | ENCRYPTION | TEMPEST |
| | HIGHER EFFECTIVE RADIATED POWER (ERP) | – |
| | SPREAD SPECTRUM | PERFORMANCE DURING JAMMING |
| | WARTIME FREQUENCIES | TEMPEST/RESISTANCE TO PROBES |
| | SIGNAL PROCESSING | REJECTION OF INVALID SIGNALS |
| HIGH ENERGY DAMAGE | OVERLOAD PROTECTION | RECOVERY TIME – MULTIPLE ATTACK SURVIVABILITY |
| | | SENSE PRESENCE OF THREAT |

Figure B-17. Test uncertainties--ECM.

## GROUND STATION ROLE IN SPACECRAFT SURVIVABILITY

The satellite ground station can have a significant role in survivability. First, the ground station may be the place where countermeasures are controlled. For instance, the satellite may sense an attack and send a warning to the ground. The ground station must determine whether the attack is real, then coordinate the appropriate response. This may include activating avoidance countermeasures employing expendables. Therefore, the attack needs to be real to preserve the expendables. Since these actions require rapid response after warning, well trained ground crews are required.

Another role of the ground station is to assess damage. This may be rather tricky because the telemetry signals may not specifically be set up to do this. Damage must be deduced from the data that comes down on housekeeping. The assessment of damage is imp[illegible]n two ways: first, the damage may indicate the level of attack (level of [illegible] second, the degree of damage will indicate the remaining operational capability. (Figure B-18)

## GROUND STATION ROLE IN S/C SURVIVABILITY

- SENSE PRESENCE AND NATURE OF ATTACK
- COORDINATE APPROPRIATE RESPONSE
- ACTIVATE AVOIDANCE TECHNIQUES
- ACTIVATE INDEPENDENT COUNTERMEASURES
- ASSESS S/C RESPONSE TO THREAT
  - DAMAGE
  - REDUCED EFFECTIVENESS

Figure B-18. Ground station role in S/C survivability.

## GROUND STATION IMPACT IN SYSTEM TESTING

An effective satellite system involves the real-time interaction of a ground station and its operators with the satellite system. The SXTF offers the potential for implementing this interaction in a controlled environment without necessarily risking the satellite. The nature of the survival response to a potential threat can be developed without using valuable flight test or real ground station assets. In addition, this approach provides a low cost method for developing, demonstrating and verifying a system survivability improvement approach in a representative realistic environment. Where necessary, fine tuning of a survivability concept can also be accomplished without incurring extensive costs.

Specific survivability features that could be examined and evaluated include (1) sensors for activation time, field-of-view characteristic, target detection capabilities, telemetry data features (2) overall system response for external attack warning time, user response timeline, data transfer requirements (3) operational personnel performance factors such as attack feature recognition time, response action procedures and attack scenario dynamics and (4) ground activated countermeasure effectiveness such as the degree of threat countering, sensitivity to threat variations, sensitivity to operator and space defense responses, sensitivity to survival system performance variations, etc. (Figure B-19)

## GROUND STATION IMPACT IN SYSTEM TESTING

- INTEGRATION POINT FOR COMPREHENSIVE DEMONSTRATION OF SURVIVAL SYSTEMS
- EVALUATE ATTACK SENSING CAPABILITY
- EVALUATE GROUND PARAMETERS WHICH ASSESS SATELLITE RESPONSE
- EVALUATE OPERATIONAL PERSONNEL AND PROVIDE TRAINING
- EVALUATE GROUND ACTIVATED COUNTERMEASURES

Figure B-19. Ground station impact in system testing.

## REPRESENTATIVE SXTF GROUND STATION SIMULATION CONFIGURATION

A preliminary concept for a ground station simulation as an integral part of the SXTF facility has been developed. The concept features a building block approach that permits development of portions of the complete system in an incremental manner. In addition the concept incorporates those specific SXTF support operations required to achieve the basic objectives of the SXTF program. (Figure B-20)

The initial step in evolving the SXTF ground station simulation capability involves selection of a computer system and associated display subsystems that are compatible with the complete ground station simulator yet will fully satisfy the basic SXTF support operations. These include such functions as control of the threat environment generation system, control of the test vehicle response and positive control of the overall test operations. In this manner, selection of the basic computer need not involve acquisition of the complete system but rather that portion required to support the basic SXTF operation, together with the appropriate "hooks" that will support capability add-on's.

The second step in the ground station evolution process could involve the addition of a capability to translate the S/C response data into representative telemetry streams. These could then be manually evaluated by S/C operators to support flight diagnostic activities. Ideally, one would like to add a simulated ground station operations center that could display the S/C responses in real time. However, this could be delayed to Step 3 if funding/schedule constraints so dictated.

Step 3 would add the real-time command data link simulation function that would give operators the ability to respond to observed S/C events in a realistic manner. This addition could incorporate a capability to translate operation responses into simulated S/C commands. The addition of these capabilities will also necessitate including a monitor function for the SXTF test controller. This will facilitate experiment control and protection of the S/C during the simulated interactive operations. Capability to include a detailed external mode simulation would not be required at this time. Rather, a manual "cookbook" system could be used to control operator response times and S/C reactions in a realistic manner. (Possibly through the use of the test controller in the loop.)

## REPRESENTATIVE SXTF GROUND STATION SIMULATION CONFIGURATION

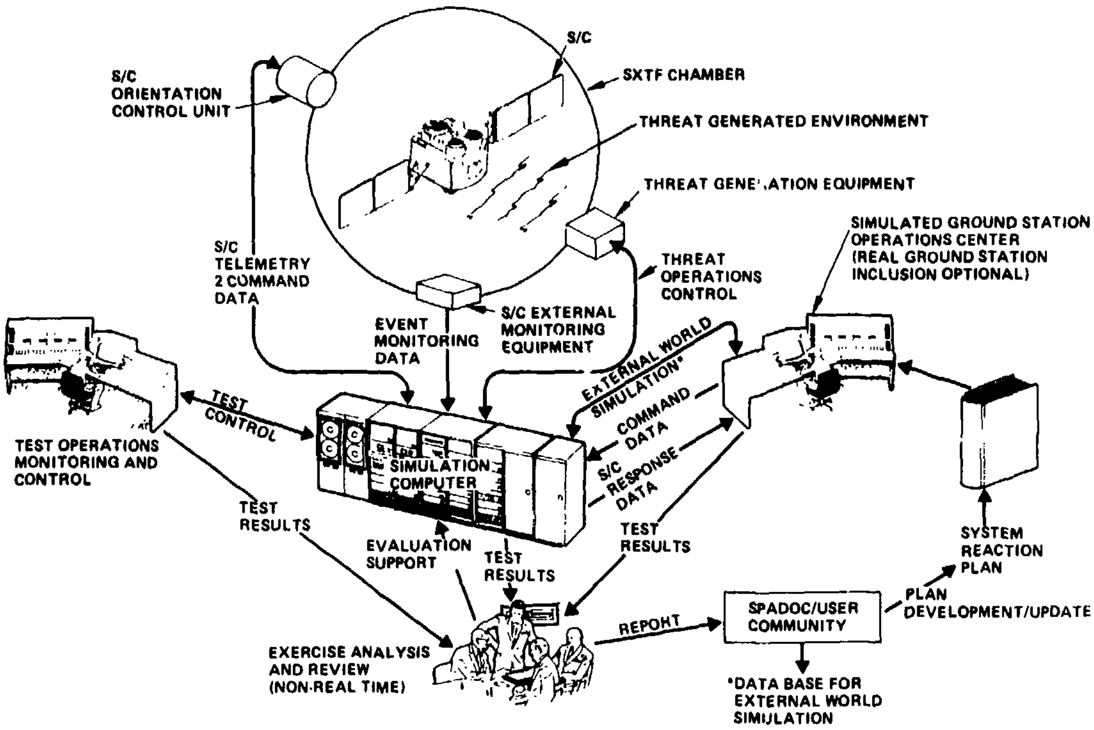



Figure B-20. Representative SXTF ground station simulation configuration.

In step 4 a detailed external world model could be added to provide for the realistic interplay between the external world space defense systems and the operators responsible for the S/C within the SXTF.

Finally, Step 5 would add the specific test analysis capability to support real-time evaluation of the test operations. This provides for the development of overall space defense operations by providing the capability to vary external world responses and integrate ground stations into the system as required.

## LASER FACILITY SETUP

It is assumed that the laser testing will be performed on a non-interference and compatible basis with the X-ray testing. Simultaneous operation of X-ray source and laser is not required. The laser test facility will therefore require only:

- Device
- Beam entry port
- Beam directing and expanding mirror
- Beam dump

The laser device can be located to one side of the adjacent high-bay area. When laser operation is required, power is switched from X-ray source. Lasant gas mixture is piped into the facility from an outside holding/distribution area. The beam entry port can be located in the loading door, introducing the laser beam into the chamber without compromising the structural integrity of the chamber or requiring any chamber modifications. A beam folding and expanding mirror (gimbaled) can be mounted immediately in front of the X-ray source. Thermal viewfactors would therefore remain virtually unaltered from the X-ray test configuration. Behind the target satellite, a beam dump can be mounted on stand-offs from the rear chamber wall. This prevents HEL irradiation of the interior chamber wall and prevents rear illumination of the target from reflections. (Figure B-21)

## LASER FACILITY SETUP

- DEVICE
  - ONE SIDE HIGH BAY
- PORT
  - IN LOADING DOOR
- POWER, LASANT GAS SUPPLY
  - NOT CO-LOCATED
- BEAM EXPANDER/DIRECTOR
  - GIMBALED AND ERECTED IN FRONT OF X-SOURCE
- BEAM DUMP
  - AGAINST BACK WALL
  - GLINTS AND REFLECTIONS HANDLED BY CHAMBER WALLS

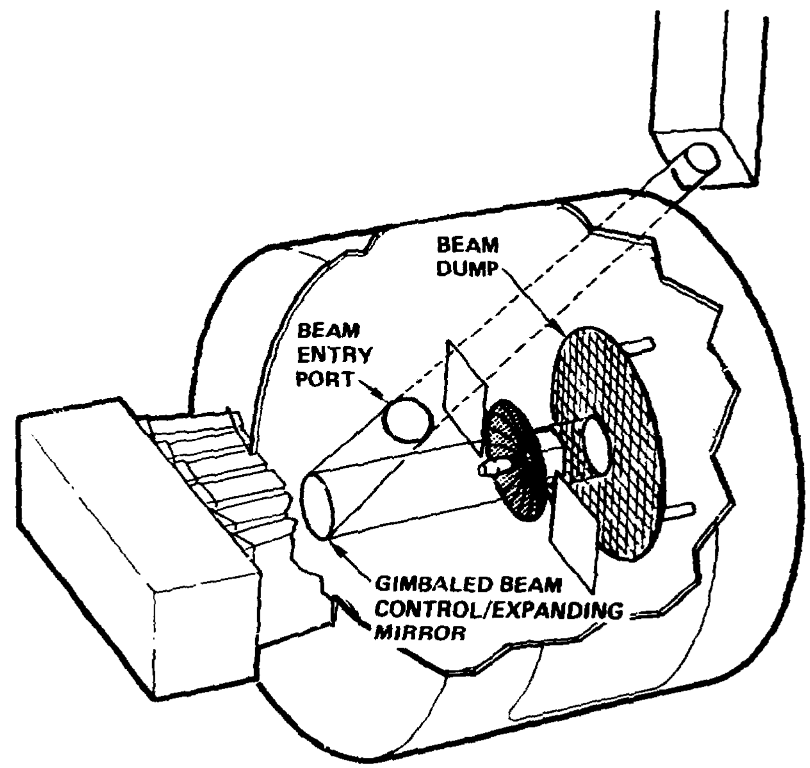



Figure B-21. Laser facility setup.

## PELLET FACILITY REQUIREMENTS

Pellet testing of actual spacecraft is probably not desirable bacause of the potential for damage. However, certain types of model testing are feasible and could be considered. For these tests, the facility would need to have fragment catchers, a pellet source and a chamber suitable for containing an explosion. These features are not readily adaptable to the SXTF requirements.

On the other hand, certain threat avoidant schemes rely on actions to be taken which can be tested. These avoidant schemes could be checked out by simulating an attack signal (flash from explosion, homing radar) and monitoring spacecraft response through telemetry. (Figure B-22)

## PELLET FACILITY REQUIREMENTS

FEATURES

- PELLET/FRAGMENT CATCHERS
- CHAMBER STRENGTH FOR CONTROLLED EXPLOSIONS
- SATELLITE DIAGNOSTICS
- PELLET SOURCE

CONCLUSIONS

- SXTF CHAMBER NOT SUITABLE FOR PELLET SIMULATION
- SXTF CHAMBER IS SUITABLE FOR SELECTED COUNTERMEASURES EVALUATION
  - GROUND STATION WARNING
  - DECOY/CHAFF DEPLOYMENT
- EW RANGE SUITABLE FOR PELLET/ASAT ECCM

Figure B-22. Pellet facility requirements.

## EW FACILITY SETUP

The facility would require a large anechoic chamber to test fully deployed spacecraft. A facility 50' x 100' might suffice. The RF source can be fairly simple since only relatively low powers are required. A computer could be used to control the source characteristic. These characteristics would probably vary from satellite to satellite particularly for investigating intelligent jamming. (Figure B-23)

Anechoic chambers are typically available at manufacturer fac es, but may not be large enough to test an entire fully deployed spacecraft.

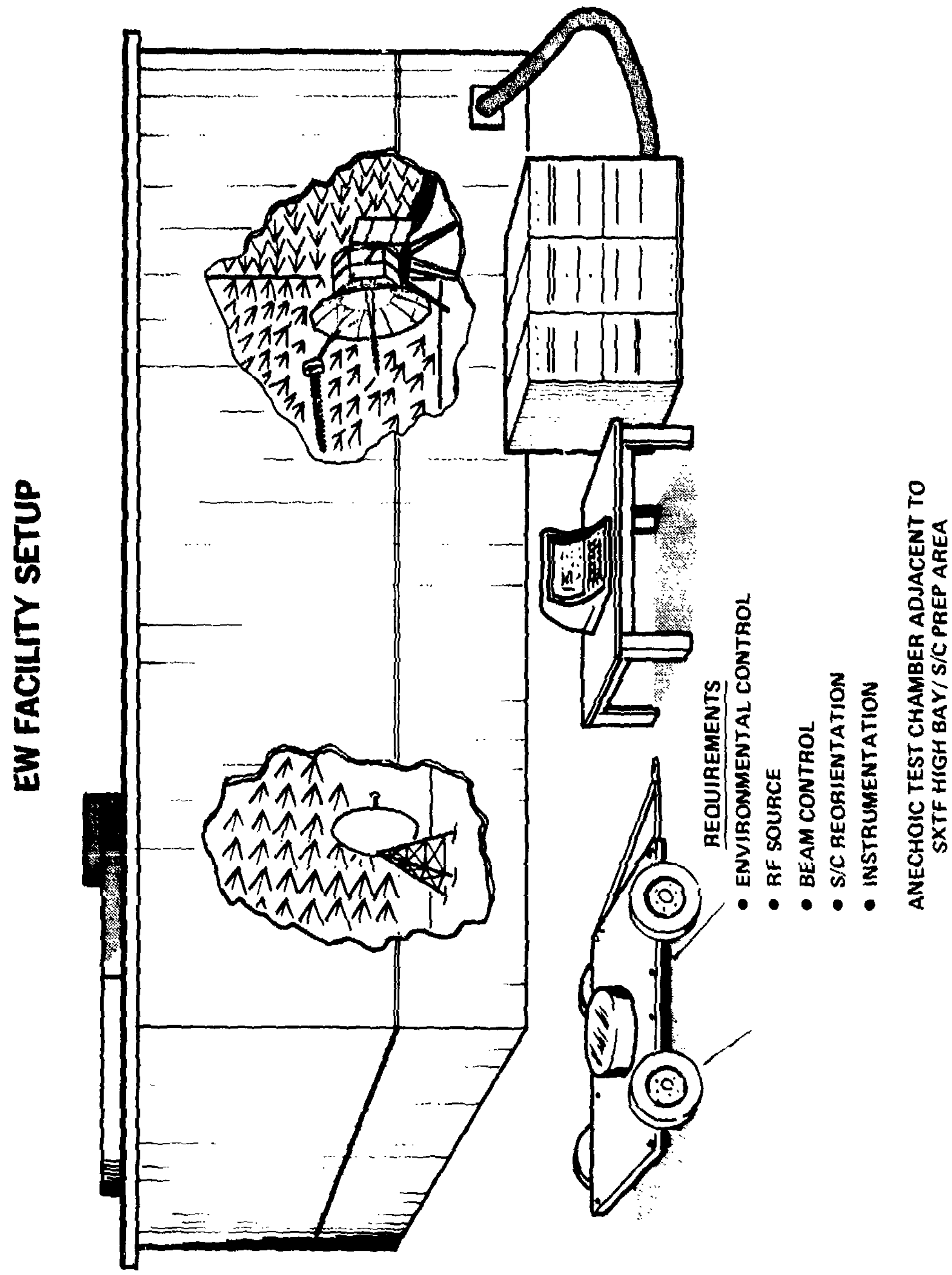



Figure B-23. EW facility setup.

## IMPACT ON SPACECRAFT PROGRAMS

The typical spacecraft program is about 2 1/2 to 3 years to production. This is a very tight schedule and typically requires a very intense qualification program. All efforts are made to minimize the time spent in qualification testing. One of the ways that qualification testing is minimized is to do as much testing as possible on engineering hardware and on subsystems in the qualification phase. These methods are reflected in this schedule for the model and component testing prior to CDR. The testing of models and components can reduce the risks associated with the weapons effects and their countermeasures. There may be developmental or technology tests of models which would be appropriate for the integrated weapons effects test facility (IWETF) in this phase.

The qualification phase is where the design is validated. Much of the validation is done at the subsystem or box level prior to full system assembly. There is very limited need for the IWETF in this phase. The final system configuration is used to validate design features which only make sense at the system level. Some of the countermeasures/weapons effects issues are system level issues. Therefore, the final phase of the qualification program could well include a trip to IWETF. (Fig. B-24)

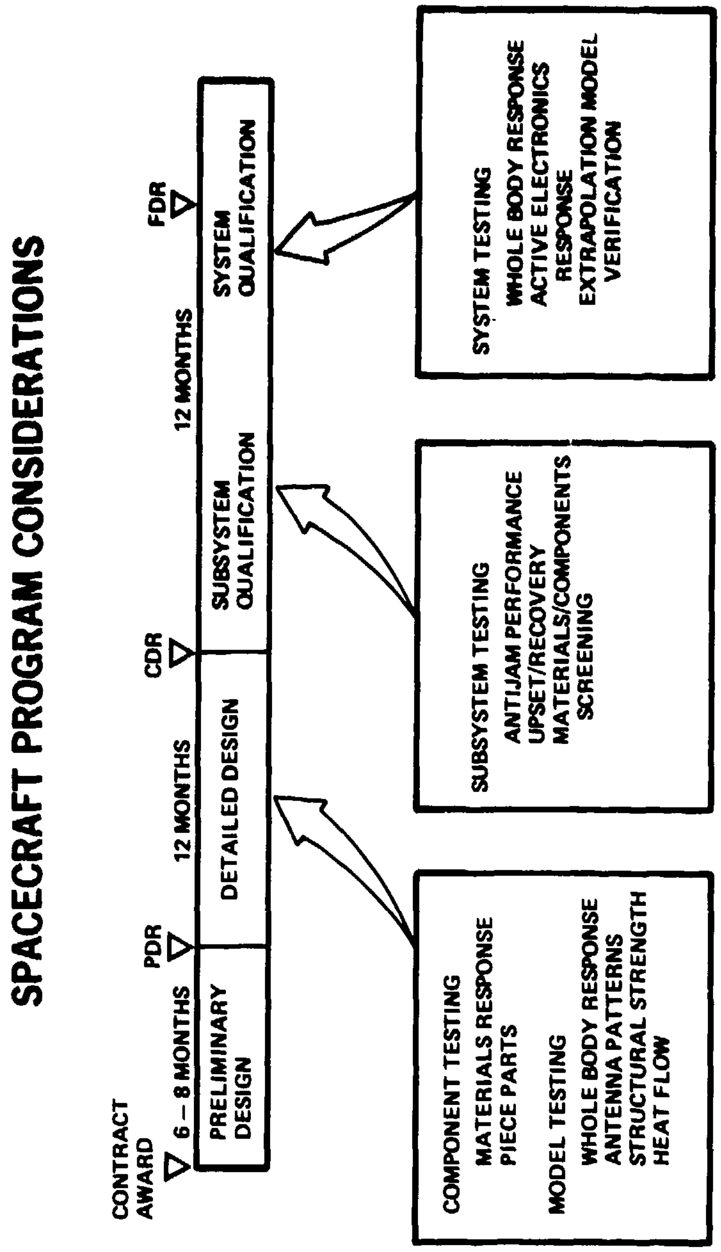


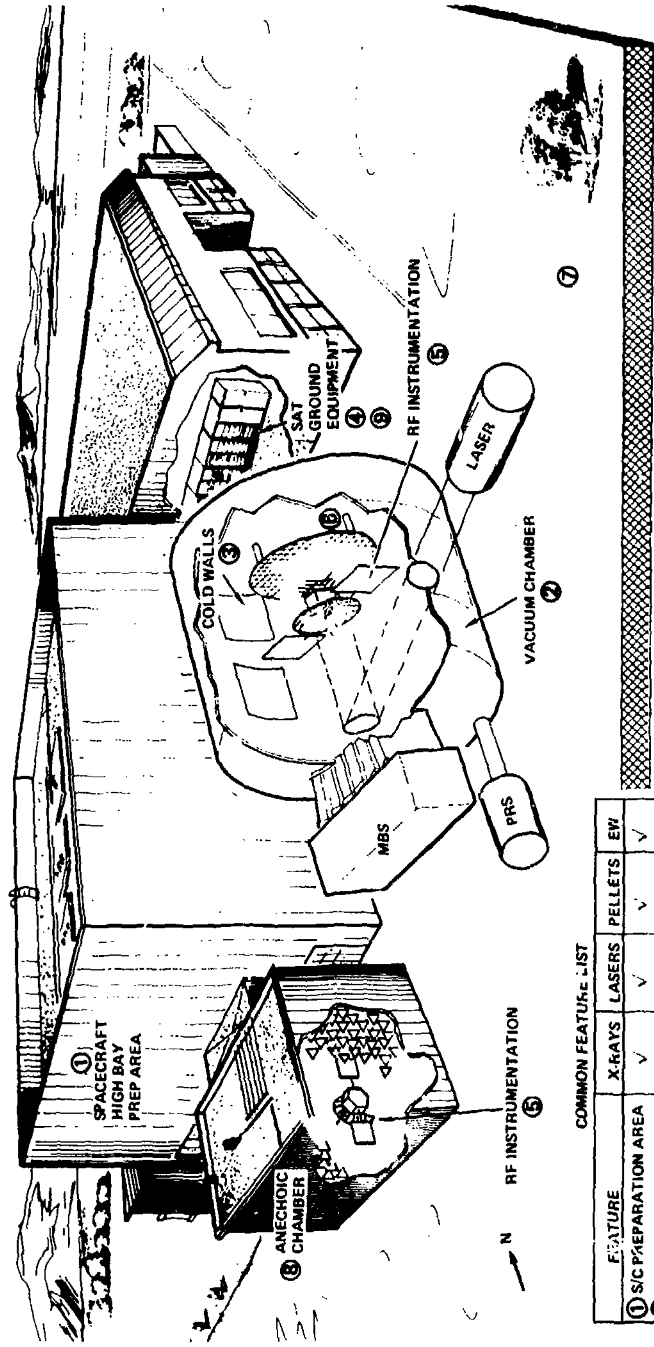

Figure B-24. Spacecraft program considerations.

## INTEGRATED WEAPONS EFFECTS TEST FACILITY

The integrated weapons effects test facility would be capable of testing x-rays, lasers and electronic warfare. In addition, a simulated satellite ground terminal would be available. One of the last elements of a spacecraft qualification program would be the weapons effects tests wherein the spacecraft countermeasures would be verified. There are many common features for the various modes of testing. Because of these common features there are significant potential cost advantages.

The basic SXTF layout with x-ray test capability is preserved. The laser tester adds very little complexity. The laser device can penetrate through an existing door and the beam dump can be a portable assembly installed and removed as required.

The electronic warfare capability requires an additional building to house the anechoic chamber. There are, however, many elements which are common with other testing such as the requirement for a satellite preparation area, satellite ground equipment and RF instruction. (Figure B-25)




COMMON FEATURE LIST

| FEATURE | X-RAYS | LASERS | PELLETS | EW |
|---|---|---|---|---|
| ① S/C PREPARATION AREA | ✓ | ✓ | ✓ | ✓ |
| ② VACUUM CHAMBER | ✓ | ✓ | | |
| ③ THERMAL CONTROL | ✓ | ✓ | | |
| ④ S/C GROUND EQUIPT | ✓ | ✓ | ✓ | ✓ |
| ⑤ RF INSTRUMENTATION | ✓ | | | ✓ |
| ⑥ DIELECTRIC MOUNT | ✓ | | | ✓ |
| ⑦ SECURE, SAFE, CONTROLLED AREAS | ✓ | ✓ | ✓ | ✓ |
| ⑧ ANECHOIC CHAMBER | | ✓* | ✓* | ✓ |
| ⑨ SATELLITE GROUND TERMINAL | ✓ | ✓ | ✓ | ✓ |

*FOR EVALUATION OF ECM/ECCM RELATED TO LASER OR PELLET ATTACK

Figure B-25. Integrated weapons effects test facility.

# APPENDIX C

# SATELLITE X-RAY TEST FACILITY
# TYPICAL USER INTERFACE DOCUMENT

TABLE OF CONTENTS

# 1.0 SCOPE

## 1.1 SCOPE

This document establishes the requirements and basic constraints imposed on the development of an architectural and engineering design for satellite operations at the Satellite X-Ray Test Facility (SXTF).

## 1.2 PURPOSE

This document defines the minimum, necessary requirements of a typical facility user to receive, checkout and install a satellite in the vacuum chamber and conduct a satellite photon exposure test.

## 2.0 APPLICABLE DOCUMENTS

### 2.1 GOVERNMENT DOCUMENTS

1. FED-STD-209B 30 May 1976
   Clean Room and Work Station Requirements, Controlled Environment

2. MIL-STD-1542 15 April 1974
   Electromagnetic Compatibility (EMC) and Grounding Requirements for Space Systems

3. MIL-P-27401C 20 January 1975
   Propellant Pressurizing Agent, Nitrogen

### 2.2 LISTING OF REFERENCES

1. MIL-STD-1246A 18 August 1967
   Product Cleanliness Levels and Contamination Control Program

2. Harry Diamond Labs
   "Wideband Analog Fiber Optics for the SXTF"
   Briefing Slides, October 1980

3. JAYCOR
   In-tank Satellite RF Links at SXTF
   200-80-256/2066 December 1980

4. JAYCOR
   Interface Control Fiber Optic Wideband Analog Data Link for SXTF
   200-80-215/2066 March 1980

5. JAYCOR
   Specifications for SXTF Fiber Optic Links
   RE-79-2066-129 April 1979

6. TRW INC
Evaluation of Candidate SXTF Sites for User Compatibility
34670-6007-UT-00 October 1980

7. TRW INC
Launch Base Test Plan, FLTSATCOM Flight Spacecraft Program
33617-600-001-01 August 1980

8. TRW INC
Spacecraft Test Planning for a System X-Ray Test
34670-6005-RU-00 May 1980

9. TRW INC
Spacecraft Considerations and Program Impacts of a Systems Level Photon Test
ALNM-7808 June 1978

# 3.0 REQUIREMENTS

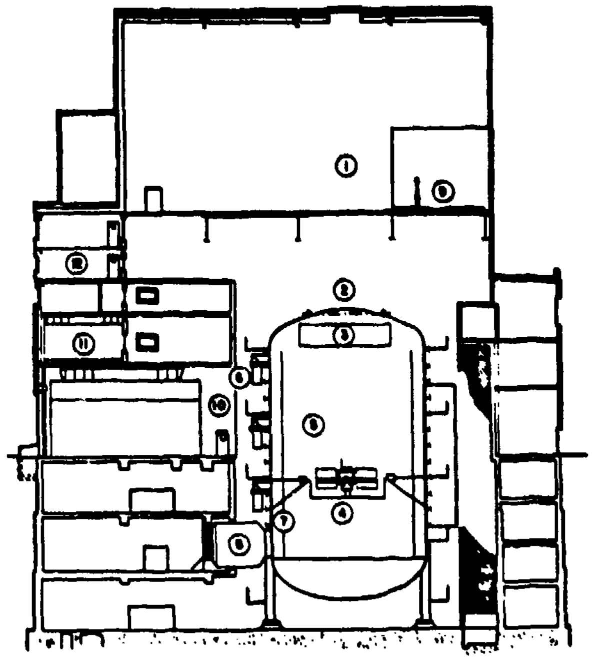
## 3.1 FACILITY DEFINITION

The SXTF at Arnold Engineering Test Center is a facility designed to test the effects of x-ray photon and electron irradiation on space satellites. The facility consists of a large vacuum chamber, two x-ray generators, electron beam charging subsystem, an ultraviolet source and ancillary equipment. The basic chamber appears in Figure C-1. The facility user is expected to be a satellite manufacturer (SM) who will demonstrate compliance with specified survivability levels to simulated threat exposures. The test article (satellite) will typically be a singular prototype. Because of the schedule constraints and prototype program status, maintenance of the satellite will be performed by user on-site engineers and technicians.

## CHARACTERISTICS

## AEDC - TULLAHOMA - MARK I CHAMBER

1 BUILDUP AREA
2 MAIN CHAMBER
3 SOLAR SIMULATOR
4 TEST ARTICLE
5 TEST ARTICLE HANDLING SYSTEM
6 DIFFUSION PUMPS
7 COLD WALL & CRYOPUMPS
8 ACCESS LOCK
9 CLEAN ROOM
10 MAIN BUILDING ENTRANCE
11 CONTROL ROOM
12 DATA ACQUISITION ROOM



- VACUUM CHAMBER SIZE: 42' DIAM x 82' HIGH (OUTSIDE)
  36' DIAM x 65' HIGH (INSIDE)
- PRESSURE ALTITUDE: SEA LEVEL TO 300 STATUTE MILES
  (1 x 10[illegible] TORR)
- THERMAL RADIATION SIMULATION: SOLAR (12' x 18');
  ALBEDO; EARTHSHINE
- WALL TEMPERATURE: 77°K (-320°F)*
- CRYOPUMP TEMPERATURES: 22°K (-423°F)**; 4°K (-452°F)***
- DYNAMIC SIMULATION: 2-SEC ZERO-G OPERATION
- PLUME TEST CAPABILITY: MAINTAIN 240,000-FT ALTITUDE
  FOR ENGINES UP TO 300-LB THRUST AND
  300,000-FT ALTITUDE FOR ENGINES UP
  TO 25-LB THRUST

*LIQUID NITROGEN, **GASEOUS HELIUM, ***LIQUID HELIUM

Figure C-1. Chamber.

### 3.1.1 SXTF Functional Flow

#### 3.1.1.1 SXTF User Test Flow

A typical test flow is depicted in Figure C-2. The test operations begin with the preparation of the spacecraft (S/C) at the manufacturer prior to shipping to SXTF. The preshipment preparations consist of installing special sensors and fiber optic transmitter/receivers and performing functional tests. In addition, other preparations will be made such as installation of test batteries to preserve the quality of the flight batteries. Spacecraft and ground support equipment (GSE) are transported to SXTF. The spacecraft is prepared for functional tests adjacent to the user screen room in the high-bay. Electronic GSE (EGSE) is installed and validated in the user screen room and initial system functional tests are performed for a baseline on the spacecraft. The spacecraft is installed in the vacuum chamber for photon exposure while in simulated orbital configuration. Procedures are verified and additional functional tests conducted. During photon exposure, data are obtained by spacecraft telemetry and the special sensors. Following the photon exposure series, the spacecraft is configured for system functional tests in the chamber and then (optionally) in the high-bay. The pre- and post-photon exposure system functional tests assess any changes in spacecraft performance. The spacecraft and GSE are then transported back to the spacecraft contractor facility. The major test activity schedule is shown in Figure C-3.

The first two weeks are for receiving, validation and calibration of the EGSE and mechanical GSE (MGSE). This is followed by a three week schedule for S/C receiving, tests and operations through final preparations for the photon exposure. The x-ray tests will be conducted over an approximate three week period. The final three weeks of the operations schedule are used for site deactivation, cleanup and return shipment of the S/C and GSE.

Receiving activities are initiated approximately fourteen (14) days prior to the scheduled arrival of the spacecraft. These initial activities include facility validations and receipt of the GSE and associated support equipment. The GSE is inspected, installed and validated in the assigned operations areas.

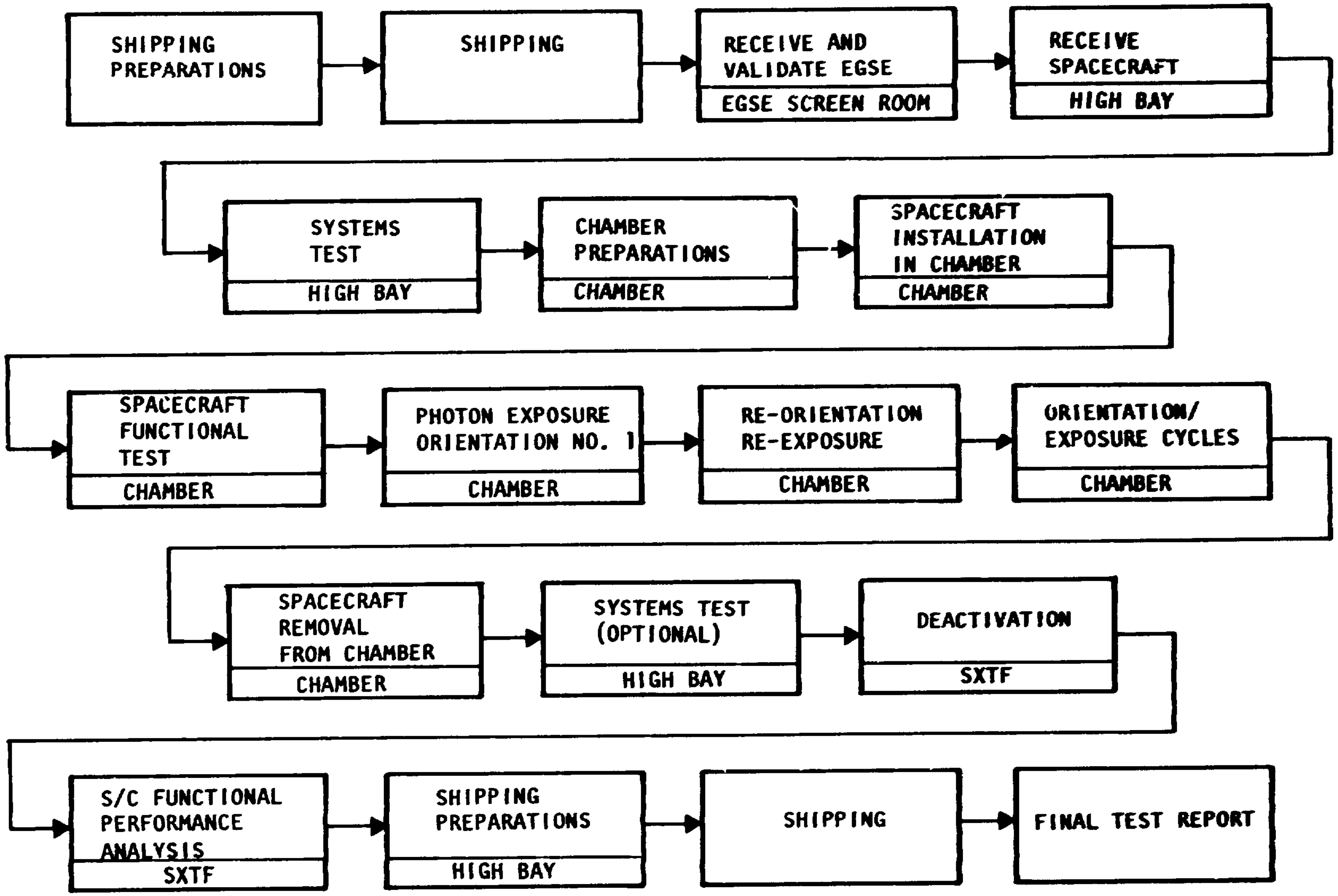



Figure C-2. SXTF task sequence.

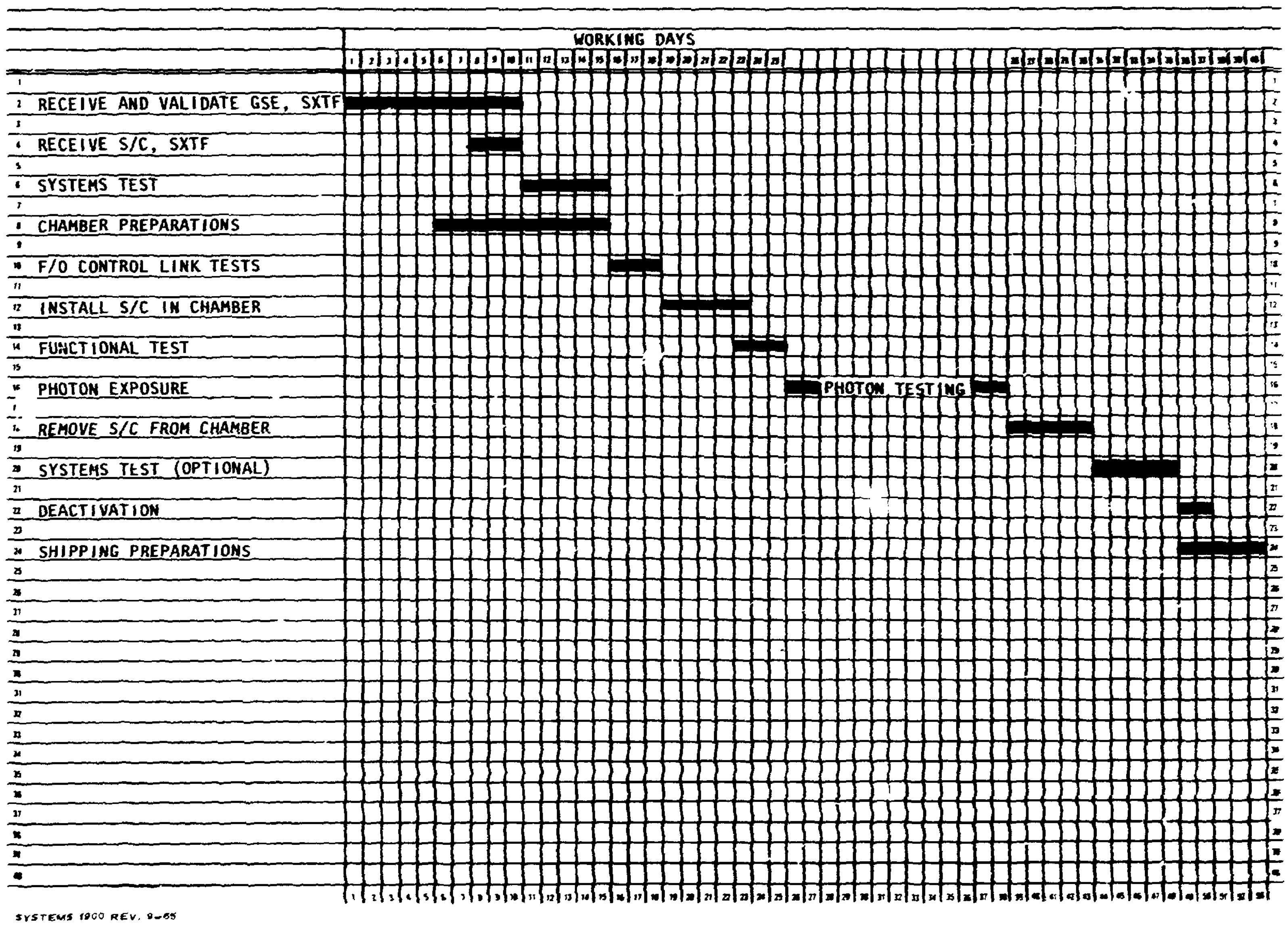



Figure C-3. SXTF test schedule.

A typical satellite is air-shipped via C5A and transported from the landing site to the AEDC, Mark I chamber. The satellite will be removed from the shipping container (see Figure C-4) and hoisted into the satellite preparation room where it will be inspected and tested prior to loading into the chamber. The test sequence described in the following i representative of a test on a communication satellite; other satellites would undergo similar tests.

A S/C systems test is performed to verify that the handling and shipping environments encountered have not degraded the functional integrity of the spacecraft. The EGSE/spacecraft will be typically configured as shown in Figure C-5, C-6, and C-7. External cooling is used for critical components, as required.

The systems test may be conducted using an EGSE battery simulator in place of flight batteries. The test batteries will remain in shorted storage during the systems test. RF testing of the communications subsystem is conducted with RF hardlines connected to the flight test couplers on the Payload Module. Following completion of the hardline portion of the systems test, the flight RF hardlines will be reinstalled and a fiber optics system will be connected. Satellite and SXTF interface tests will be performed to validate the system. The shorts on the test batteries will be removed and the batteries recharged to full capacity.

Following systems tests the spacecraft will be lowered into the chamber where it will be configured (see Figure C-8) and positioned for the photon exposure. Following installation in the chamber, an electrical and RF interface check will be performed to verify compatibility between the spacecraft and associated EGSE and to verify the hardline links and the RF (fiber optics) link between the control center and the spacecraft. A typical test set/spacecraft configuration is shown in Figure C-9.

Following verification of the fiber optic control link performance, the EGSE-S/C hardlines are removed and the EMP sensor fiber optic links tested. During these tests the S/C will be in the standby mode using battery power. Facility provided test pulsers will be used to excite the SGEMP sensors and the resultant data will be recorded in the facility data collection screen room.

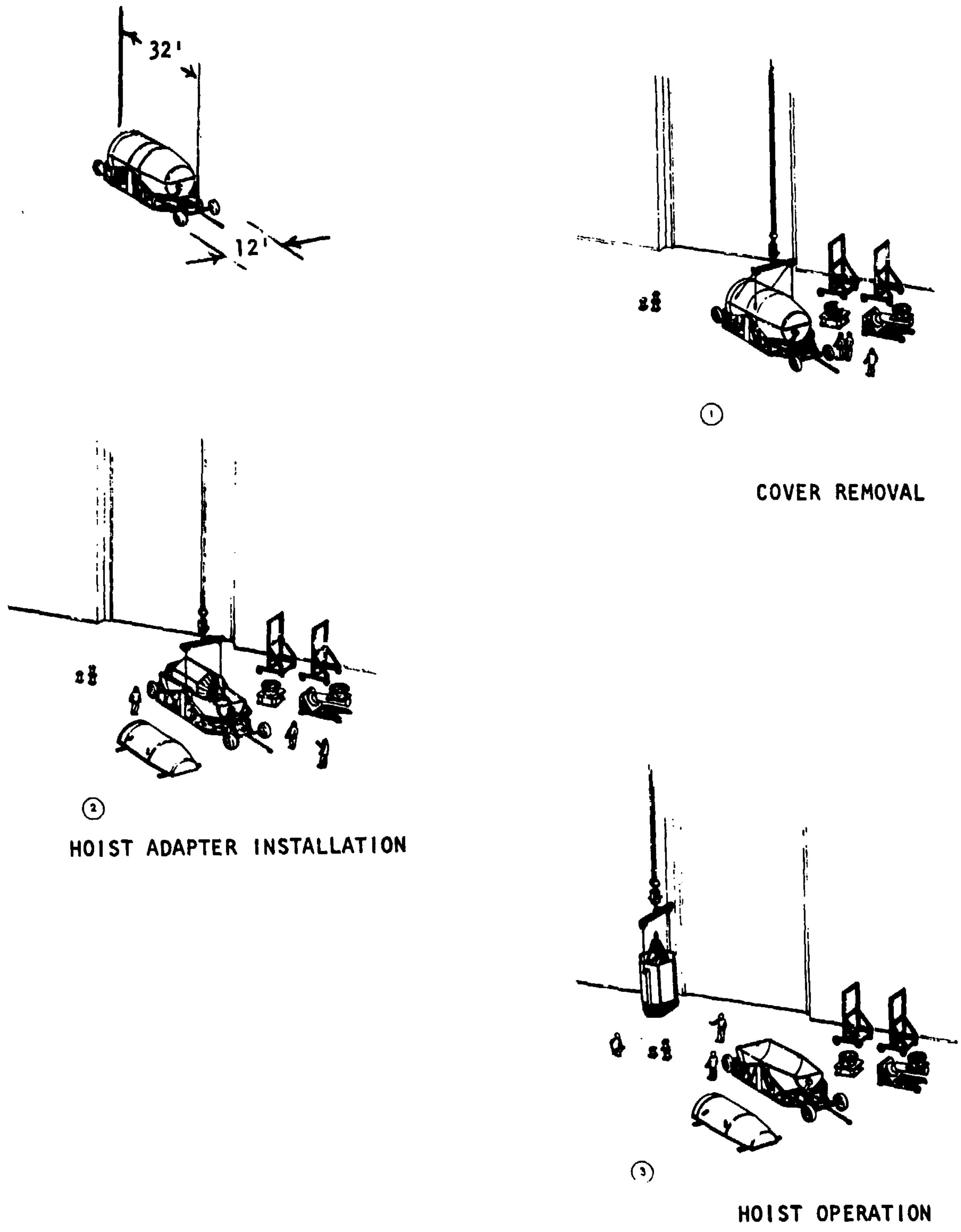
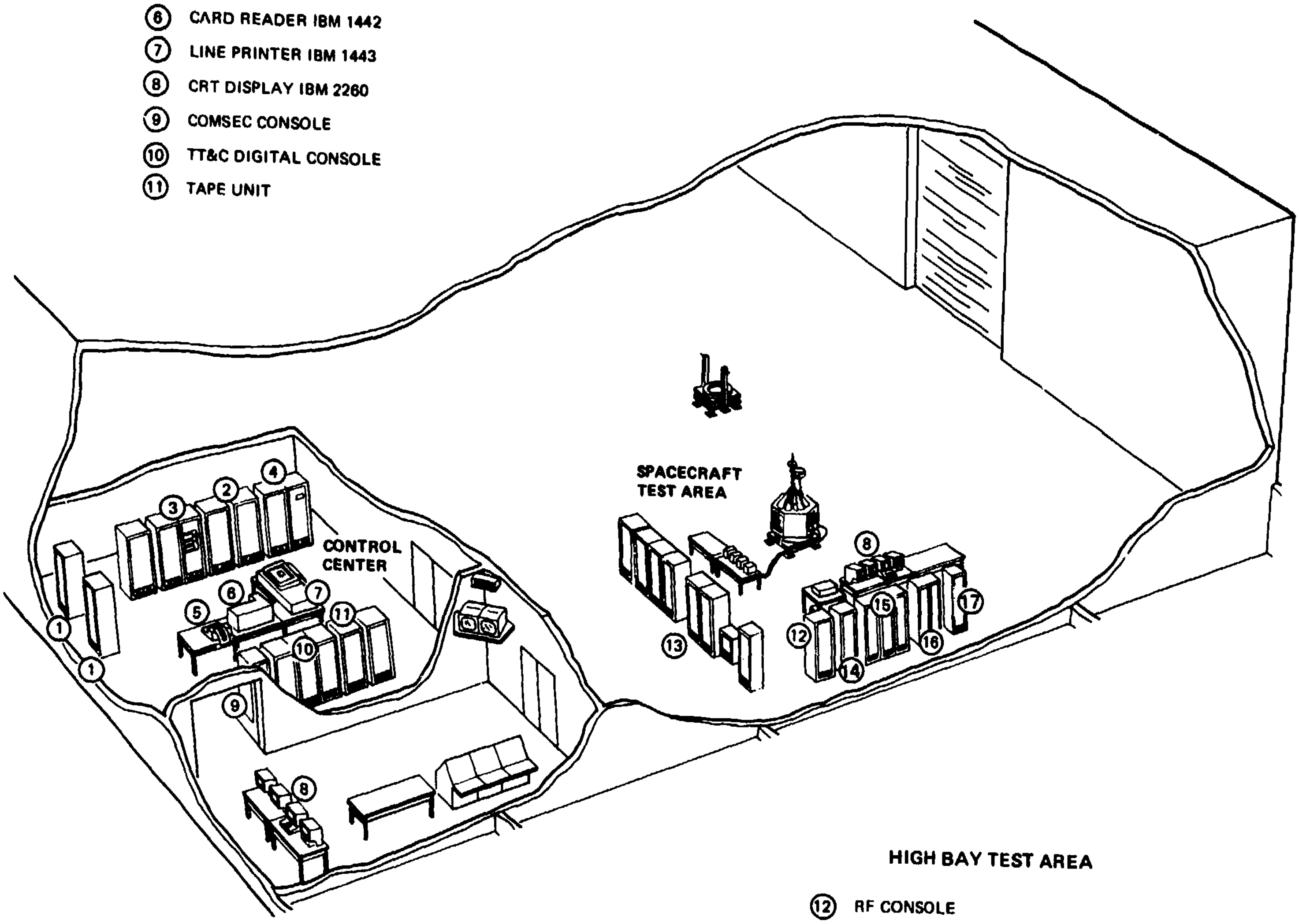



Figure C-4. Transporter receiving area operations.

**CONTROL CENTER**

1. DISC UNIT IBM 1810
2. DATA ADAPTER UNIT IBM 1826
3. CENTRAL PROCESSOR IBM 1801
4. DISPLAY CONTROL UNIT IBM 2848
5. TYPEWRITER IBM 1816
6. CARD READER IBM 1442
7. LINE PRINTER IBM 1443
8. CRT DISPLAY IBM 2260
9. COMSEC CONSOLE
10. TT&C DIGITAL CONSOLE
11. TAPE UNIT




**HIGH BAY TEST AREA**

12. RF CONSOLE
13. COMMUNICATIONS CONSOLES
14. ORDNANCE AND TEST POINT MONITOR
15. CONTROLS SUBSET
16. POWER SUBSET
17. BATTERY SIMULATOR
18. TRICKLE CHARGER

Figure C-5. Typical high bay test area.

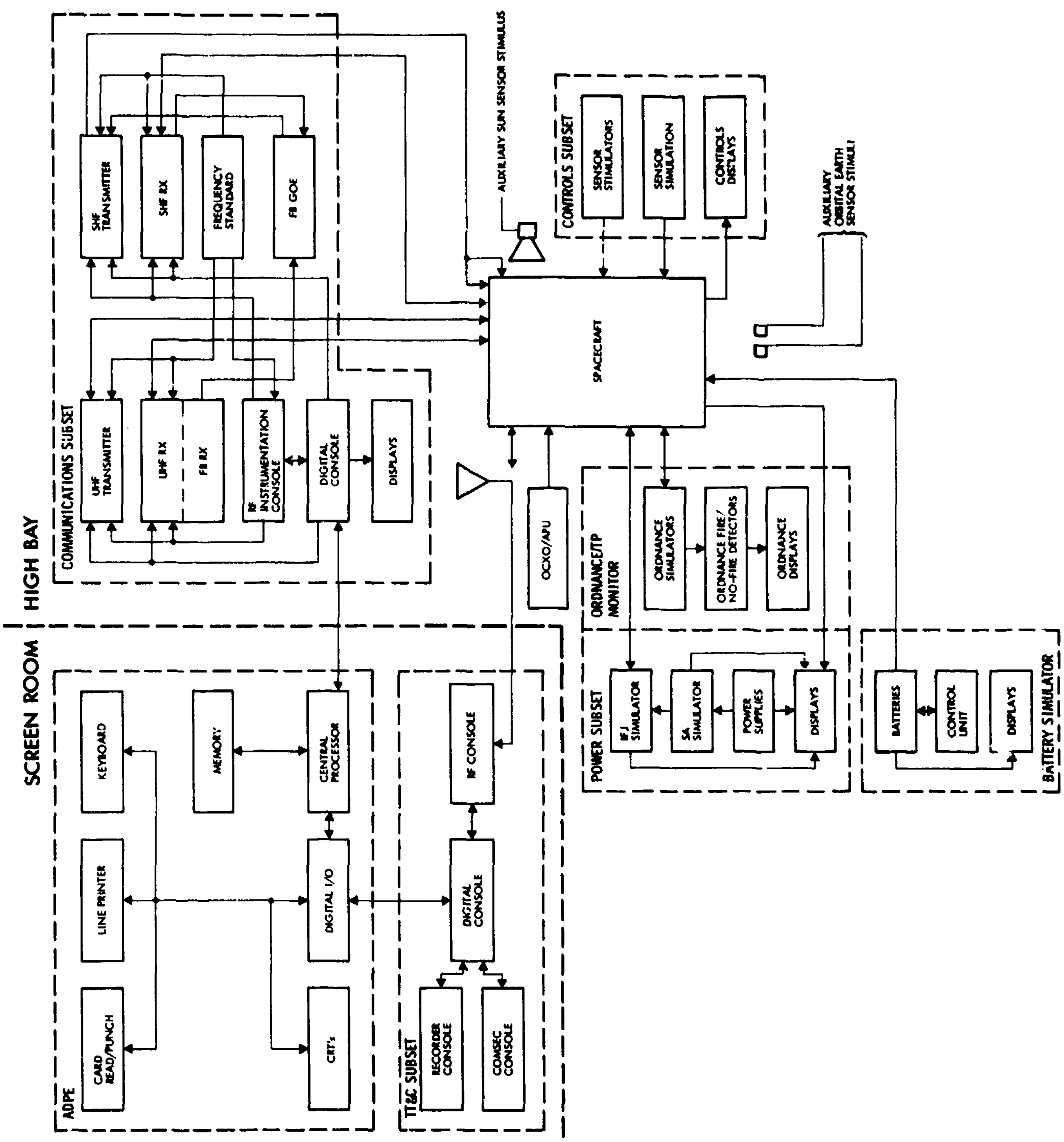
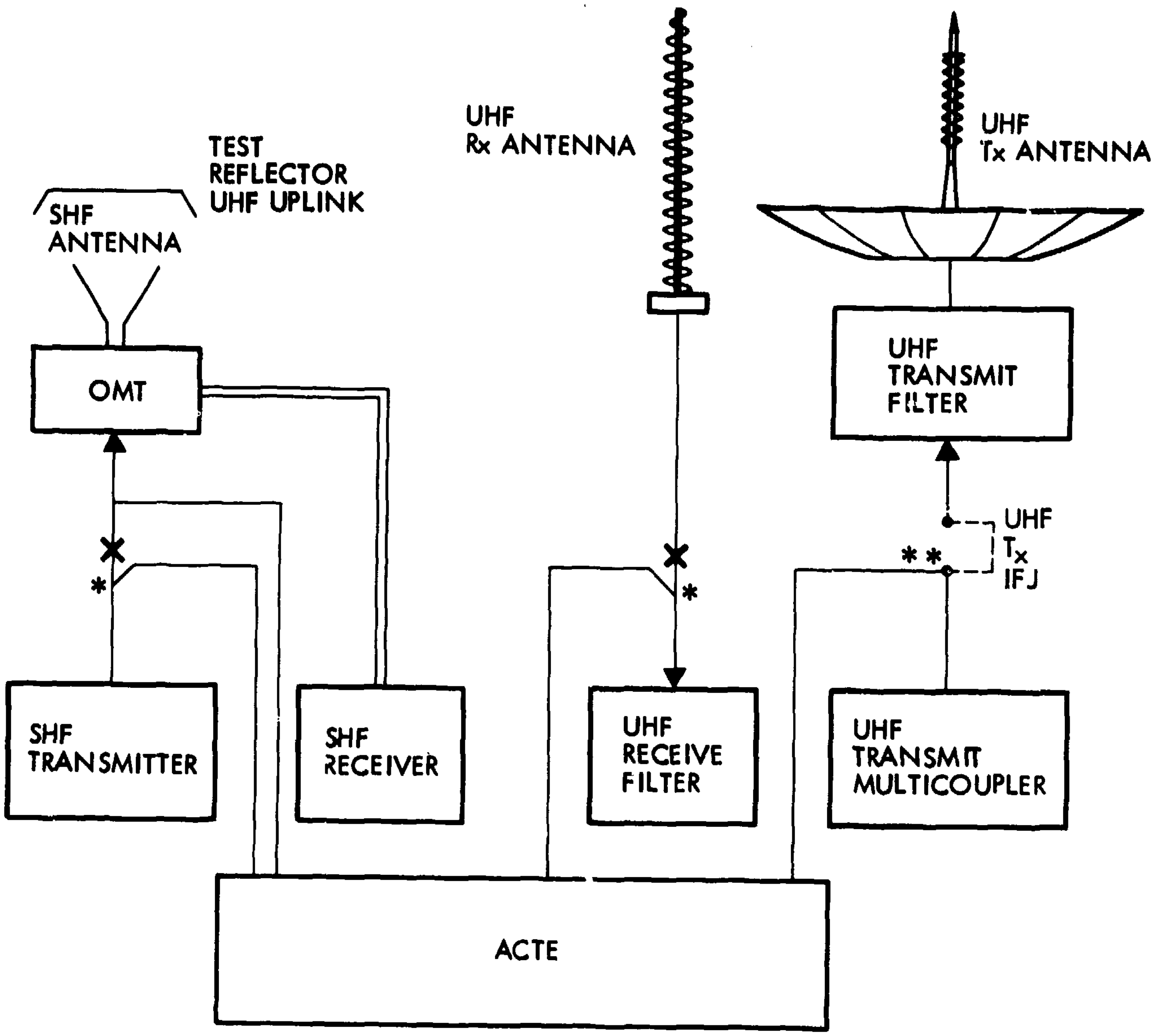



Figure C-6. System test block diagram.

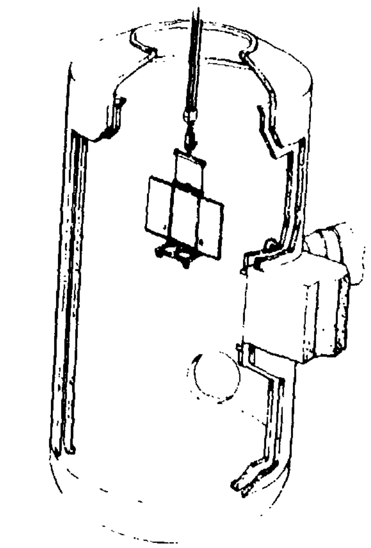



NOTE:

* RF HARDLINE TEST CONNECTION. PARTIAL SOLAR PANEL MOVEMENT REQUIRED FOR ACCESS.

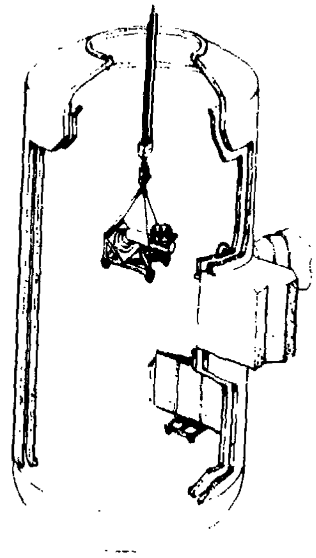
** RF HARDLINE TEST CONNECTION AT +X LOCATION ON P/L MODULE.

Figure C-7. Communications subsystem systems test hardline interface.



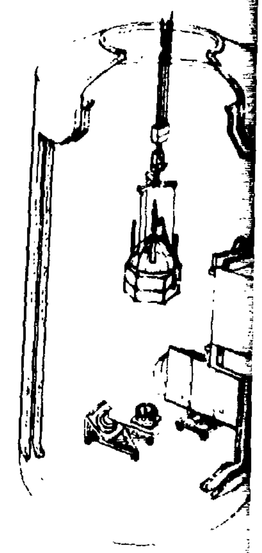
(1)

LOWERING SOLAR PANEL



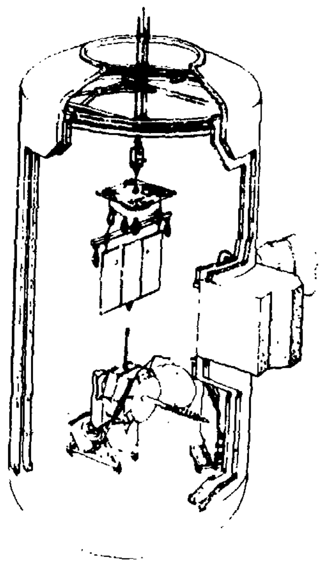
[illegible]

LOWERING ASSEMBLY FIXTURE



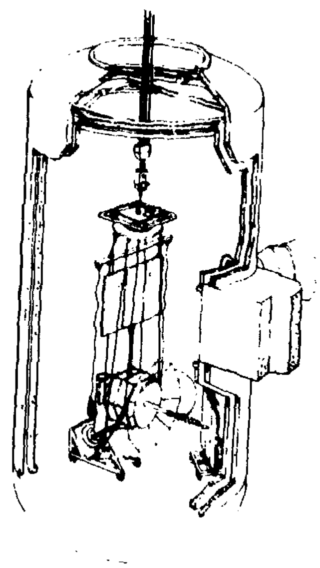
(3)

LOWERING SPACECRA



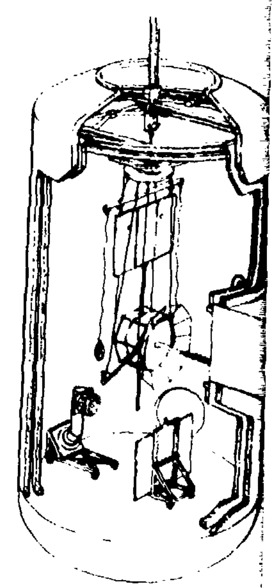
7)

LOWER SOLAR PANEL
AND STRONGBACK



8)

ATTA ...PER
SUSPENSION LINES



9)

PARTIAL LIFT

1

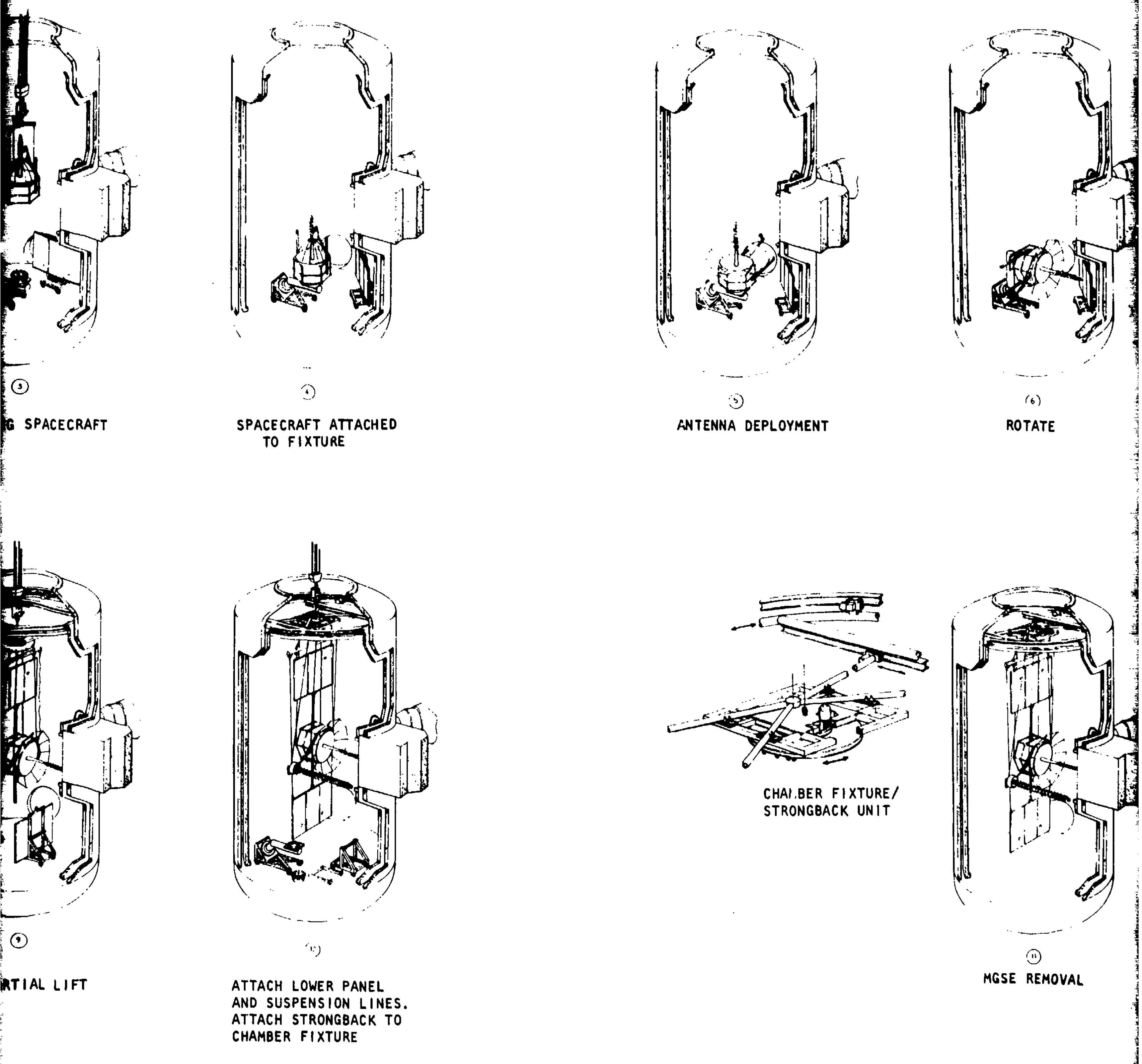



Figure C-8. Spacecraft chamber installation.

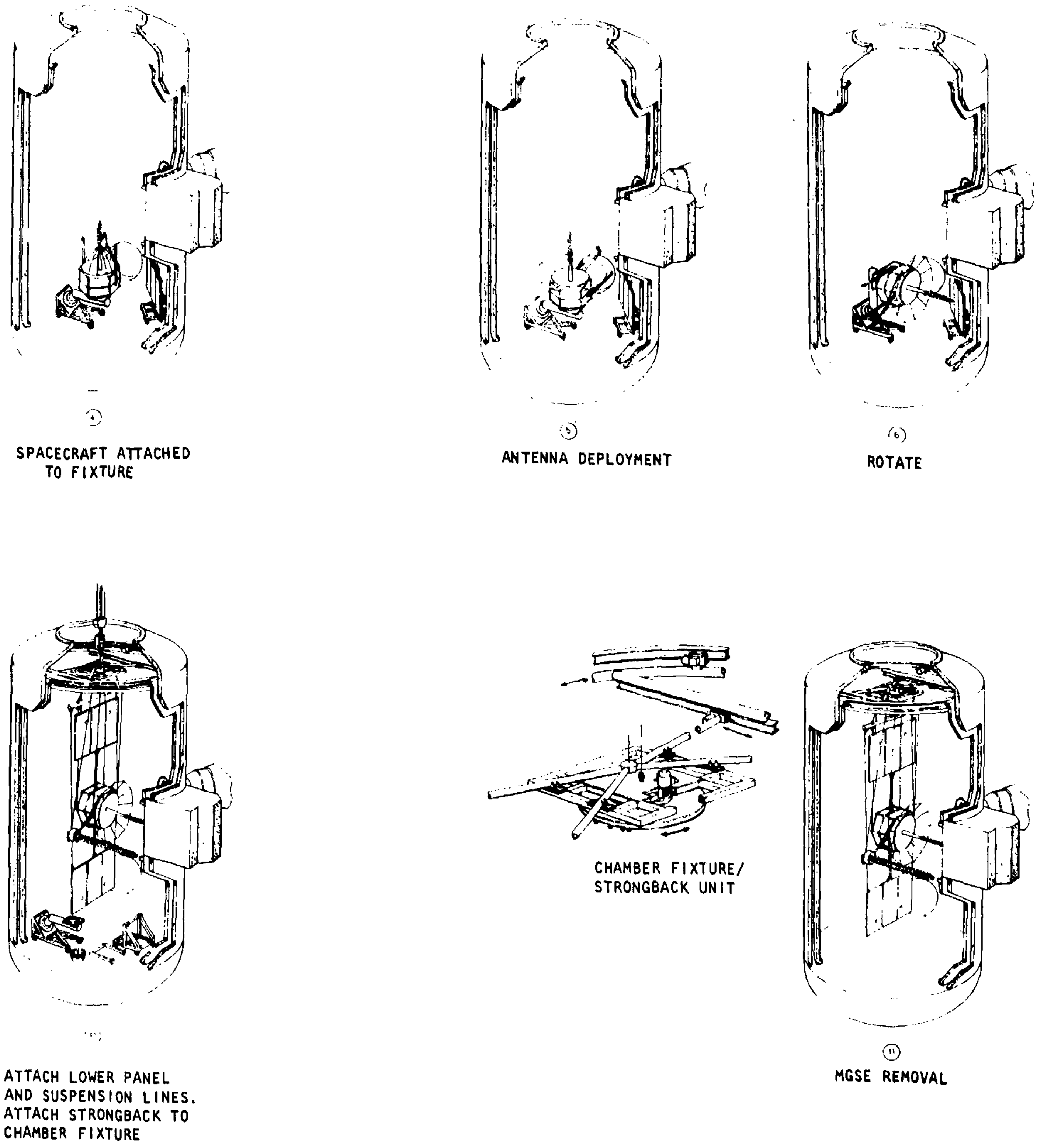



Figure C-8. Spacecraft chamber installation.

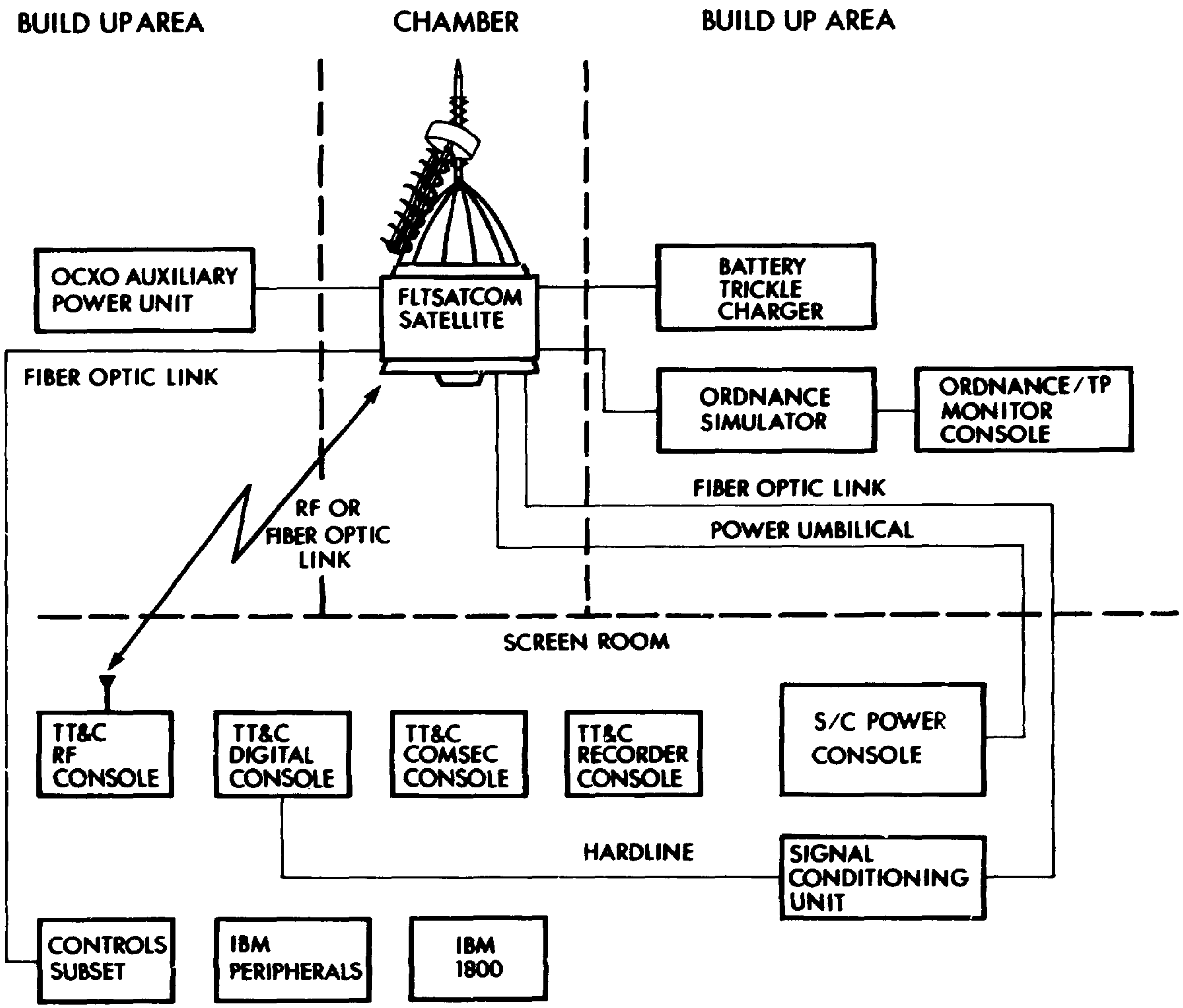



Figure C-9. Chamber interface tests.

A practice countdown will be performed prior to final closeout of the chamber to provide operator training and procedure validation. This test will include the following functions:

- Test cable removal, in-flight jumper connection and remote umbilical operation.
- Thermal configuration verification
- Electrical configuration status verification
- Umbilical retraction/insertion procedure verification
- Telemetry status monitoring
- Spacecraft re-orientation procedure verification
- Chamber closeout

Following chamber closeout and pump down, the S/C power will be brought up during cold wall fill. S/C functional performance will be verified once thermal equilibrium is reached. The S/C power umbilical will be retracted in preparation for the photon exposure.

The facility pulsers will be fired and data collected from both SGEMP sensors and telemetry monitoring. After each exposure the S/C power umbilical will be inserted and batteries recharged. Various spacecraft operational modes and orientations will be used during the tests as detailed in the test plan. Functional tests will be conducted as required. At test completion the S/C will be powered down in conjunction with chamber warm-up.

Following pressurization of the chamber, the chamber will be opened. The S/C will be removed in reverse order from the loading procedure. The S/C will be configured in the preparation area for additional testing, if required, or pre-shipment preparations. Upon completion of preparations the S/C will be returned to the S/C contractor facility.

3.1.2 User - Facility Interfaces

3.1.2.1 Functional Interface - Figure C-10 depicts the user-facility organizational functional interfaces: The facility user will operate or monitor use of facility equipment to move or manipulate the spacecraft. An SGEMP data analyst station will be used for quick-look of test data and validity estimation. A thermal monitoring station will be used to monitor chamber temperature especially during chamber cooling or warming transitions; these data will be evaluated with respect to the spacecraft temperatures to avoid over or under heating. The facility status will be monitored to coordinate test events, assure environmental control including cleanliness and to establish daily facility support requirements as defined in the test plan. Figure C-11 illustrates the facility support provisions.

3.1.2.2 Physical Interface - Figure C-12 depicts the user-facility physical interfaces: Test force, spacecraft with supporting equipment, EGSE, MGSE, spares and transportation equipment. These interfaces are supported through functional areas as presented in Figure C-13.

3.1.2.2.1 Administrative Area - The user test force will be supported out of an administrative area. This area provides office space for administrative and engineering user test force staff.

3.1.2.2.2 Spacecraft - The spacecraft will interface with the controlled facility environment, hoisting cranes, fiber optic links for test data and spacecraft control, and chamber RF antennas. Facility cranes will be used to deliver the spacecraft between the receiving area-high bay area and high bay-chamber. Facility provided fiber optic links will be used for collection of test data (analog links) and spacecraft control (digital links). Chamber RF links will be used to transmit and receive spacecraft data and commands. The spacecraft is supported by a user provided suspension system. The suspension system also supports the fiber optic cables and the system interfaces to the facility strongback. The facility strongback can be manipulated to position the spacecraft in azimuth for photon exposure tests. The suspension system also supports a user provided umbilical mechanism. The zero entry umbilical interfaces to a facility provided umbilical connector and associated facility wiring into the screen room.

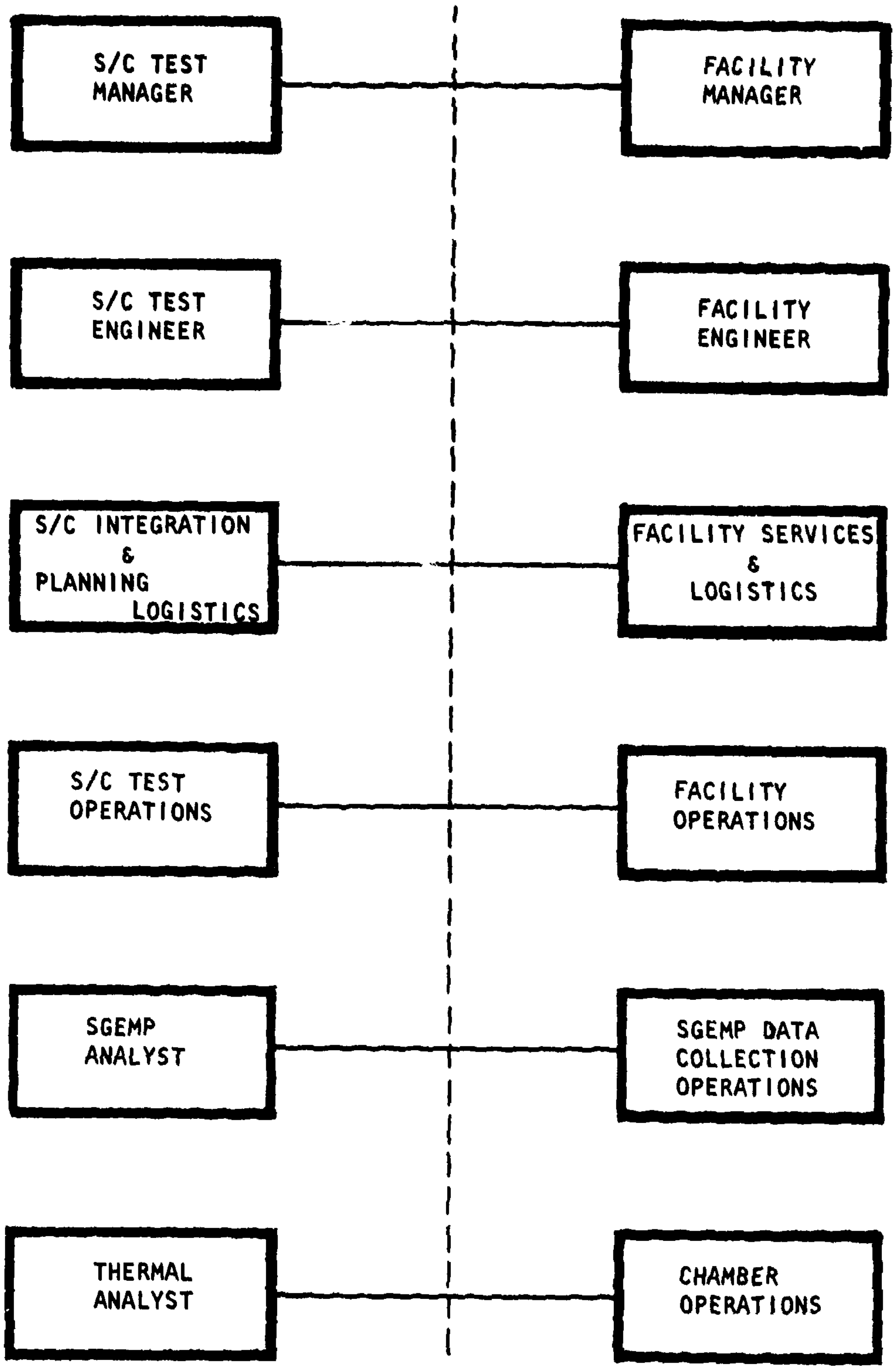



Figure C-10. User/facility organizational interfaces.

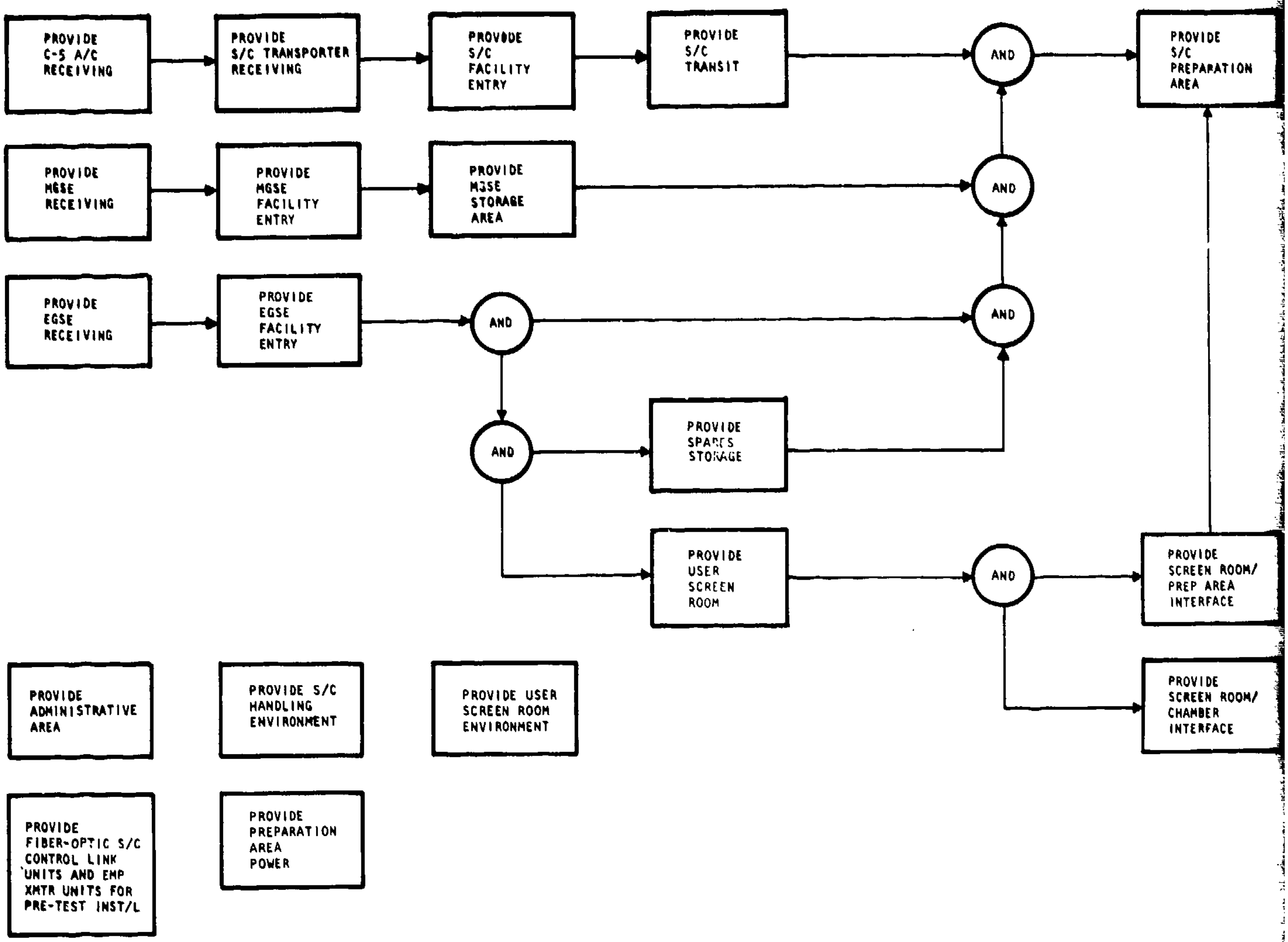
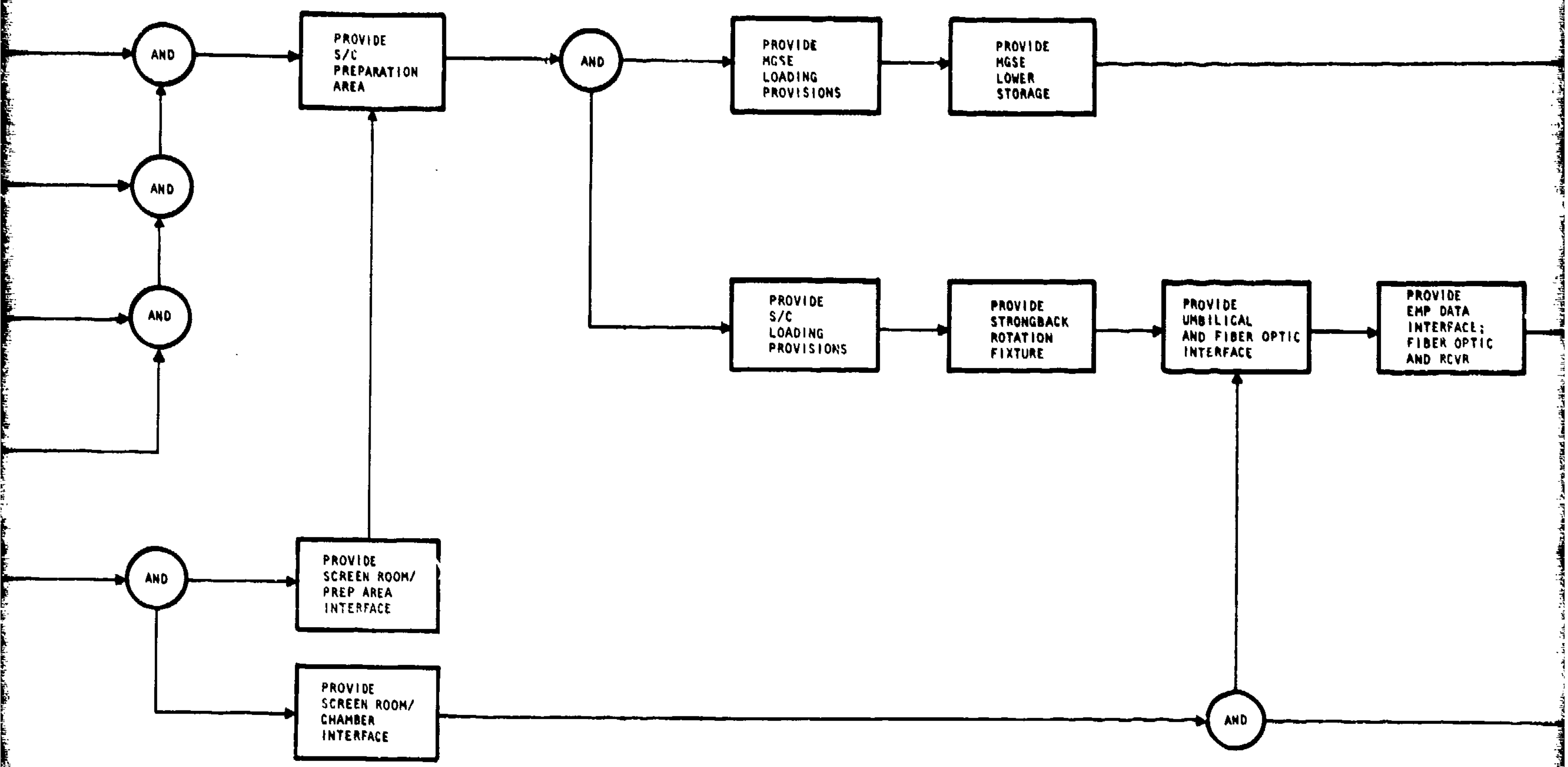
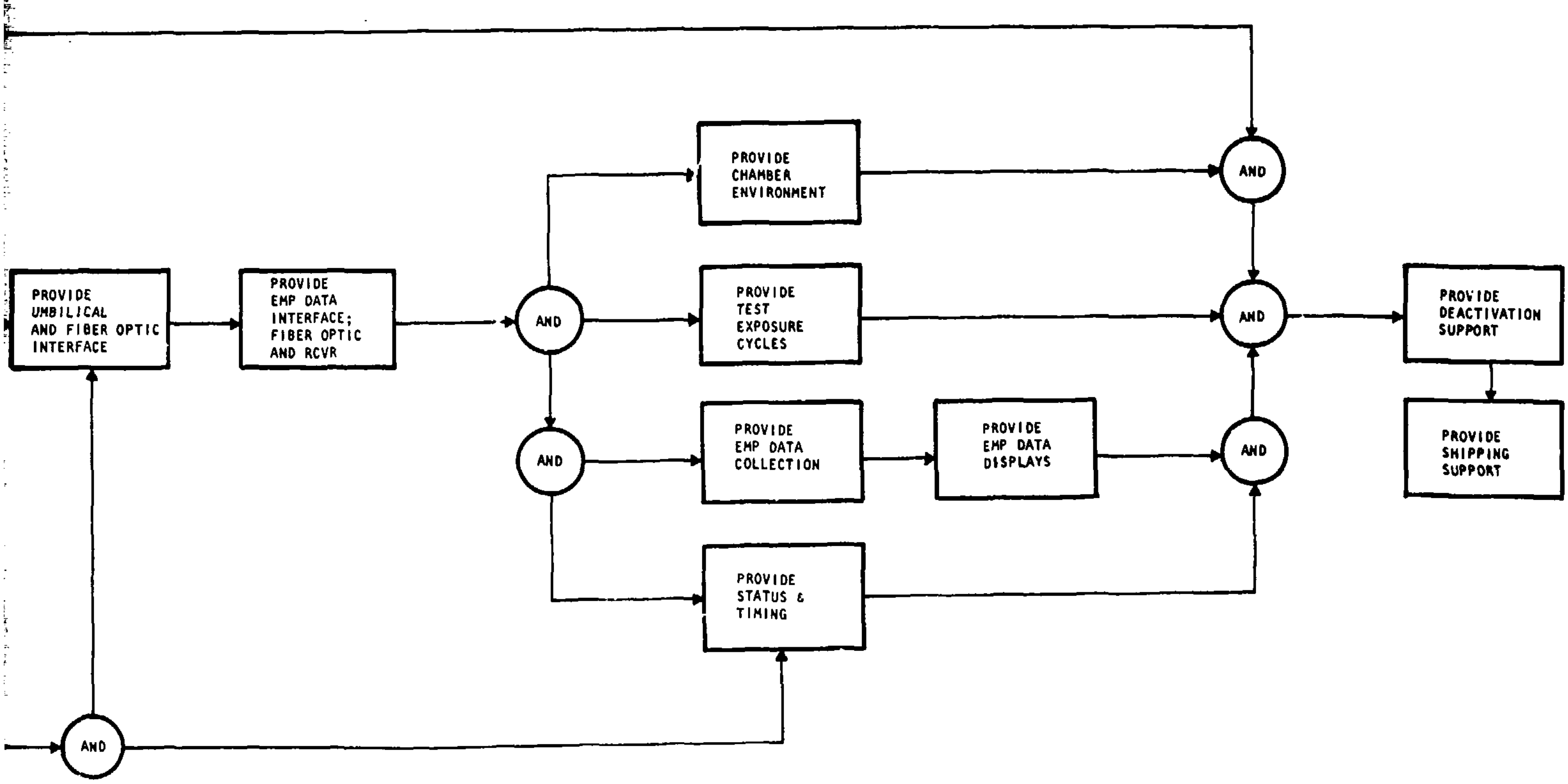



Figure C-11. Facility support provisions.


AND
AND
AND
PROVIDE S/C PREPARATION AREA
AND
PROVIDE MGSE LOADING PROVISIONS
PROVIDE MGSE LOWER STORAGE
PROVIDE S/C LOADING PROVISIONS
PROVIDE STRONGBACK ROTATION FIXTURE
PROVIDE UMBILICAL AND FIBER OPTIC INTERFACE
PROVIDE EMP DATA INTERFACE; FIBER OPTIC AND RCVR
AND
PROVIDE SCREEN ROOM/ PREP AREA INTERFACE
PROVIDE SCREEN ROOM/ CHAMBER INTERFACE
AND

2


PROVIDE UMBILICAL AND FIBER OPTIC INTERFACE
PROVIDE EMP DATA INTERFACE; FIBER OPTIC AND RCVR
AND
PROVIDE CHAMBER ENVIRONMENT
PROVIDE TEST EXPOSURE CYCLES
PROVIDE EMP DATA COLLECTION
PROVIDE EMP DATA DISPLAYS
PROVIDE STATUS & TIMING
PROVIDE DEACTIVATION SUPPORT
PROVIDE SHIPPING SUPPORT

3

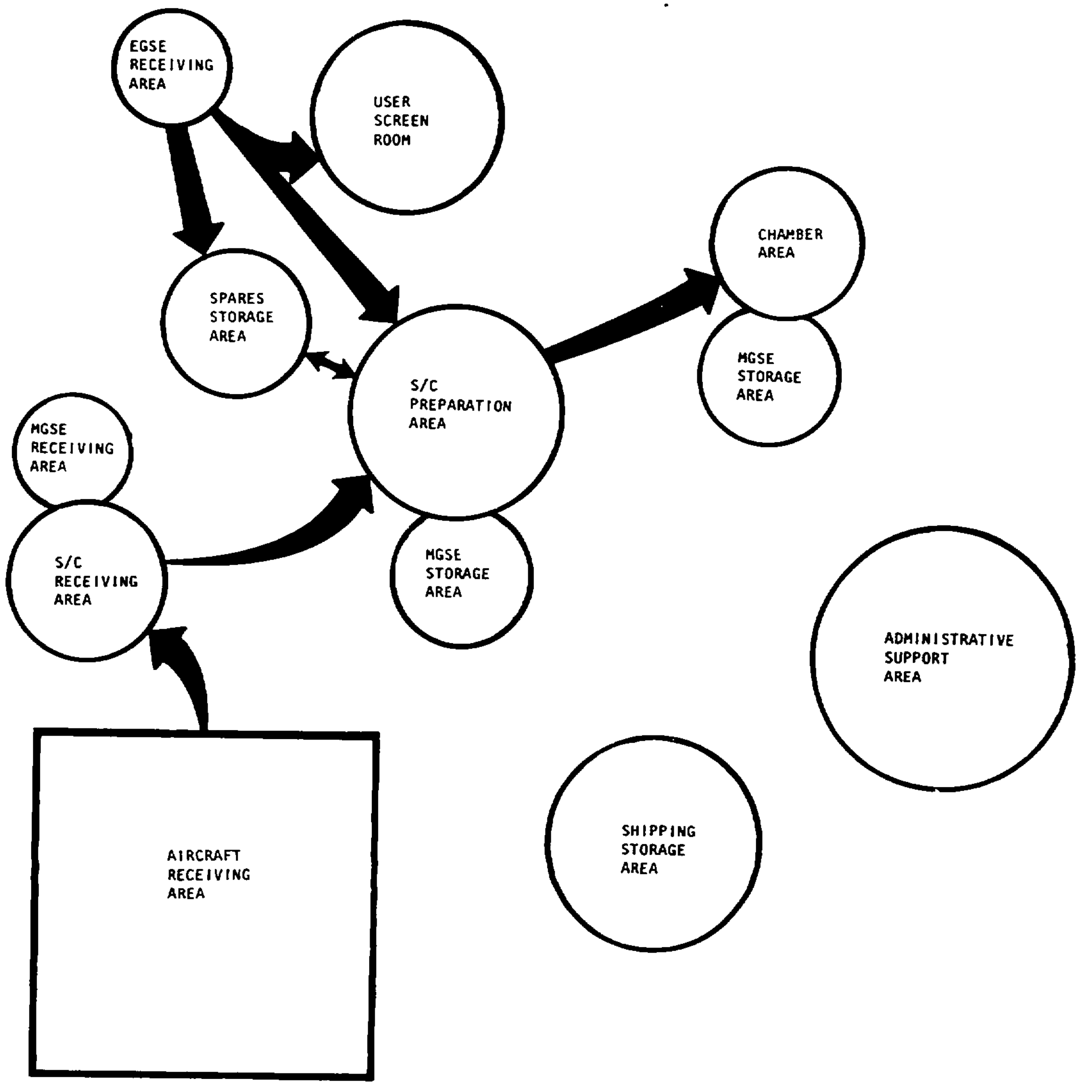



Figure C-13. Physical interface facility areas.

3.1.2.2.3 EGSE - EGSE will be installed in the User Screen Room and in the Spacecraft Checkout area. In each location the EGSE will interface with the facility power and environmental conditioning. Screen room EGSE will collect signals via hard line and facility provided digital fiber optical control links. Hard lines will be routed through a facility provided screen room disconnect to the Spacecraft Checkout area, the chamber umbilical and chamber RF antennas. Facility cable trays will be used as a conduit for signal cables.

3.1.2.2.4 MGSE - MGSE will be used during spacecraft assembly and check-out operations. When not in use the MGSE will be stored near the operation areas.

3.1.2.2.5 Spares - Spare equipment will be stored in facility provided storage areas.

3.1.2.2.6 Transportation Equipment - Packing material, the S/C transporter and transporter support instrumentation trailer will be stored in facility provided storage areas.

## 3.2 CHARACTERISTICS

### 3.2.1 General

Requirements for each functional area will be described in the following sections. These sections describe requirements which are common to several functional areas and are associated with the movement of spacecraft, MGSE and EGSE. MIL-STD-1574 should be used as a design guide for all facility interfaces involving S/C handling.

3.2.1.1 Cranes - A mobile crane shall be provided to remove the cover from the S/C transporter. A dripless overhead crane shall be provided to lift the S/C from the transporter and position it in the preparation area above the chamber. The same (or another) crane shall be used to lower, position and transition the S/C onto the strongback rotation fixture. The crane capacity shall be sufficient to lift approximately four tons (three ton S/C without panels, suspension lines, strongback, etc.). Crane shall be equipped with dual suspended interlocking controls to permit control at both lower and upper lift extremes (both vestibule and chamber). Crane accelerations shall be limited to ± 2.5g hoist and ± 2.0g traverse.

3.2.1.2 Spacecraft Cleaniness Environment - Prototype test article spacecraft have typical contamination requirements under MIL-STD-1246A to hold particulates to level 300 and non-volatile residue to level A. Generally, the spacecraft manufacturer meets these requirements by cleaning the spacecraft. In order to minimize contamination, the facility shall provide a clean environment in spacecraft handling areas. A clean environment may be provided by good housekeeping practices and controlling airborne particulates to better than 100,000 per FED-STD-209.

3.2.1.3 Spacecraft Thermal Environment - Typical test article spacecraft are designed to function properly over a limited temperature range. Radiative cooling and local heaters are used to maintain the spacecraft thermal balance in the cold of space. These thermal conditions are difficult to achieve during ground testing without the use of special equipment and a dependence on a steady test area temperature. Accordingly, the temperature in all areas where the powered or non-powered spacecraft is handled shall be adjustable between 65-75°F, controlled to ± 3°F. The relative humidity (RH) shall be less than 50% with excursions permitted to 70% RH for less than one hour.

3.2.1.4 Spacecraft Thermal Monitoring - Out of tolerance environment may result in later test failures and mechanical alignment problems. The facility shall provide temperature and humidity recording with hardcopy data available upon request. The buildup area and chamber shall be instrumented. Chamber instrumentation may be portable as required.

3.2.1.5 Gaseous Nitrogen - The transporter receiving area, buildup area and chamber shall be provided with access to gaseous nitrogen for purging. Nitrogen quality at user outlets shall meet MIL-P-2740C, Type 1, Grade A except for moisture which shall be Grade C or better. Outlet pressure shall be regulated to 20 $\pm$ 5 PSI. Total flow will not exceed 30 SCFH.

3.2.1.6 Security - Some candidate test S/C contain classified equipment; others are, in addition, visually classified. Classification level of Secret is expected to be the highest level. The facility shall provide controlled access to spacecraft handling areas. Data transfers between the S/C and EGSE may also be classified (encrypted and clear text) and the ADPE will be processing classified data. TEMPEST security provisions should be provided.

3.2.1.7 MGSE/EGSE Floor Loading - Areas supporting EGSE transit, use and storage shall have structure sufficient to support EGSE wheeled rack units with a base of 40x70 inches weighing one ton. Transport elevators used for EGSE/MGSE shall be a minimum of 8 (high)x8 (wide)x10 (deep) feet with a load capacity sufficient for 2 tons plus four technicans. The elevator door must be 8x8 feet.

*Load requirements are based on largest MGSE/EGSE from the DSP and FLTSATCOM programs. Other MGSE may be larger (say 12 feet diameter) but weigh less. Large MGSE will be loaded by crane rather than elevator.*

3.2.1.8 Facility Illumination - Spacecraft handling areas (lifting, testing, service areas) shall be provided with sufficient illumination to permit close, precise work. Illumination of 100 foot candles in the work zone has generally been sufficient.

3.2.2 Administrative Area

a. Civil.

The administrative area shall have lighted parking for 50 standard vehicles. Special fencing and security are not required.

*The 50-vehicle requirement is based on an approximate test force size of 75 and the need to accommodate government test monitors, associate contractors and others at planning meetings.*

b. Architectural.

(1) The area shall house a test force of up to 40 persons for a normal eight hour day. The area shall house up to 8 persons for overtime operations during the remaining 16 hours. Test planning conferences for up to 40 people shall be accommodated in a conference room. Typical user test force manning appears in Figure C-14. An area flow diagram appears in Figure C-15.

*The test force was sized using DSP and FLTSATCOM personnel requirements and deleting those positions necessary for launch operations and propellant/explosive operations. Positions were added to cope with the expected mechanical complexities associated with spacecraft suspension and to handle the SGEMP data.*

(2) Doors entering the administrative area shall be provided with cypher locks (or equivalent) for security.

c. Electrical.

(1) Standard three wire 115 VAC power shall be provided via standard outlets in each office.

(2) Separate circuit 115 VAC power shall be provided for a copy machine (20A) and for a coffee machine (15A).

(3) Lighting intensity shall be 100 foot candles minimum at desk level in working areas and 50 foot candles in hallways.

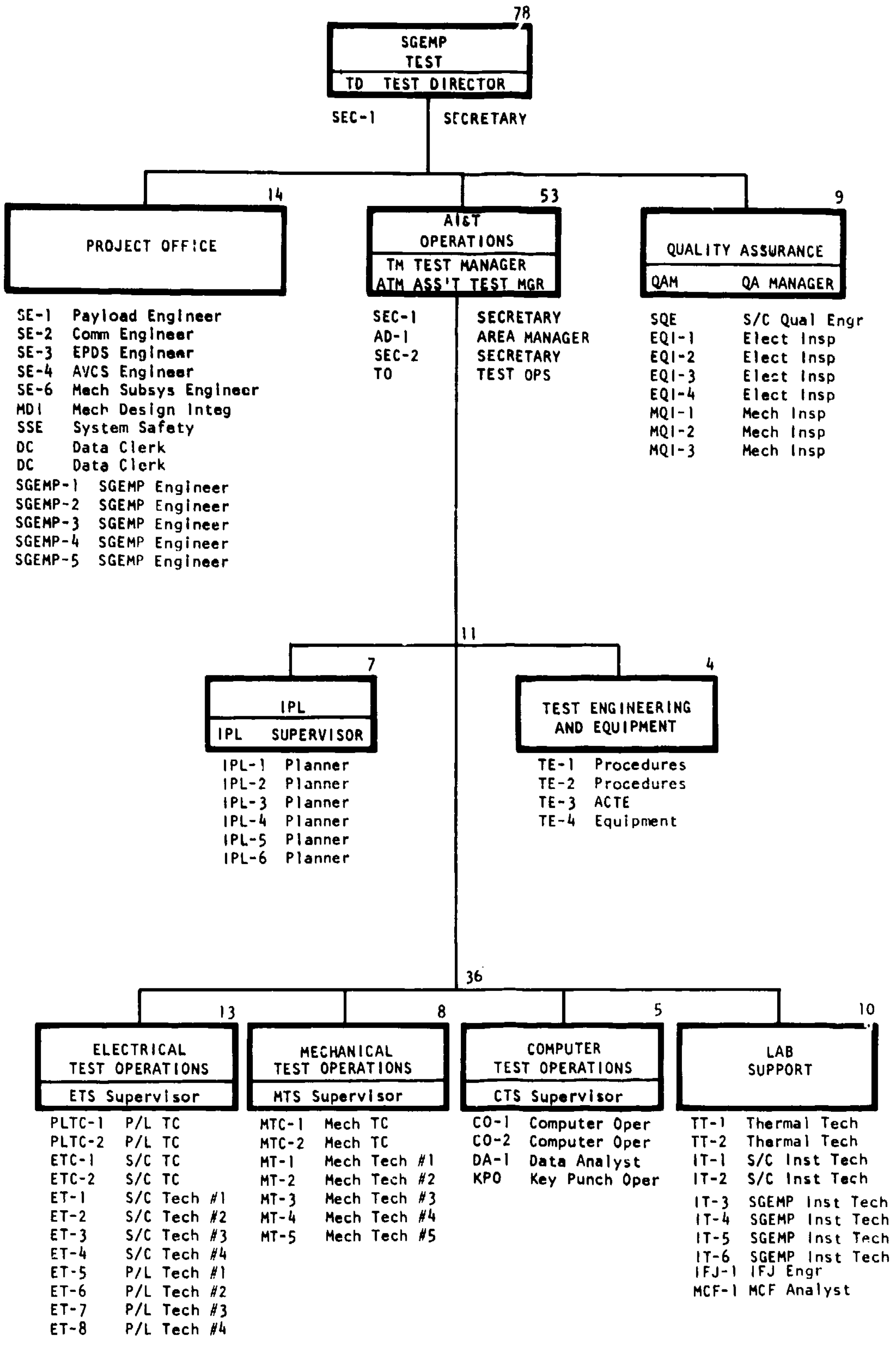



Figure C-14. User test force.

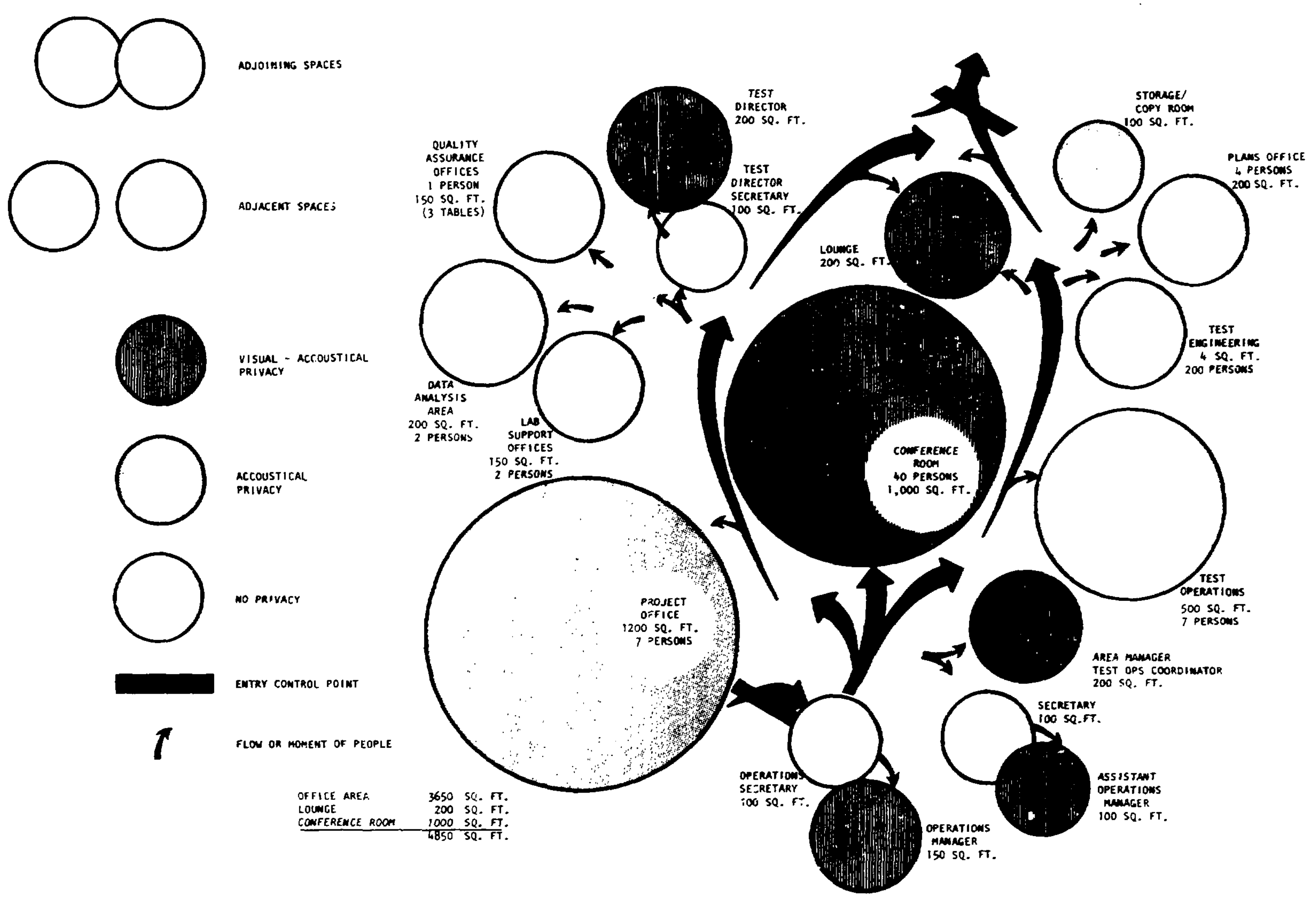



Figure C-15. Administrative area flow.

(4) Two direct telephone lines to commercial trunk service are required. Two intra-base telephone lines are required. A telephone shall be provided in each office and in the conference room. A multiple telephone shall be provided near each secretary area for call director use with intercom connect to each office. Two telephones shall be provided in the data analysis area.

### 3.2.3 Receiving Areas

The receiving areas will be used to support staging and off load of the S/C, MGSE and EGSE as outlined in 3.1.1.

a. Civil.

Entry roads into the cargo off load area shall accommodate a 12 ton standard flatbed tractor trailer and tractor. Maneuvering room shall be provided to permit entry of transporter into transporter loading dock. A typical convoy is shown in Figure C-16.

*Requirements based on locating the spacecraft transporter (30 ft. long x 12.5 ft. wide x 12.5 ft. high and 9 tons with tire loading of 68.5 PSI).*

b. Architectural.

(1) Area subdivisions are associated with equipment and work flow differences. The spacecraft will be removed from the transporter and hoisted into the high bay assembly area. Mechanical GSE will be similarly hoisted into the high bay. Electronic GSE will be offloaded onto a loading dock, unpacked, and moved into the high bay/screen room area via elevator.

(2) Entry door. Exclusive of crane clearance height, the high bay entry door shall be a minimum of 16 feet high and 16 feet wide in order to permit entry and movement of the transporter.

*Requirements based on transporter length and width. Maximum MGSE dimensions were 12 feet.*

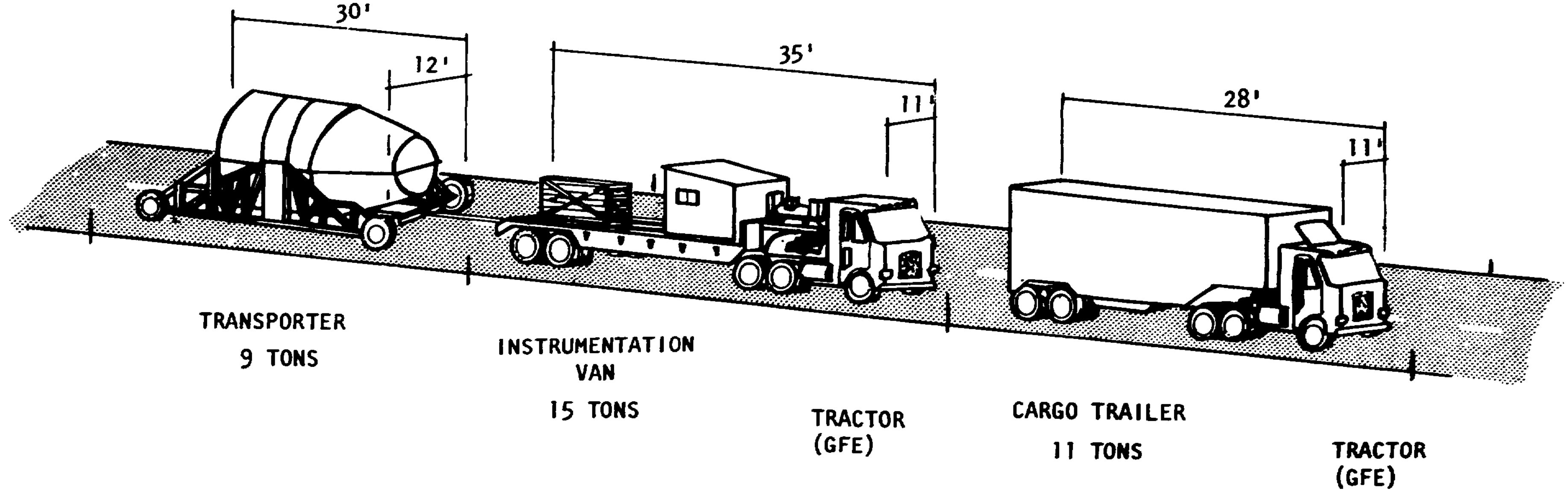



Figure C-16. Spacecraft convoy.

(3) EGSE entry doors. Doors shall be a minimum of 8 feet tall and 8 feet wide to permit entry of EGSE with dimensions 40Wx108Lx90H inches.

*Requirements are based on largest EGSE rack which has dimensions of 40 inches width x 108 length and 90 height.*

(4) EGSE Loading Dock.

Provide a 15-foot deep sheltered loading dock for EGSE off-loading and unpacking. Dock to be located near EGSE elevator entry. Dock length of about 30 feet to permit temporary storage of packing material and to permit forklift maneuvering.

(5) Transporter Loading Area.

Provide a sheltered vestibule for removal of transporter cover and spacecraft. Loading area shall permit direct off load of spacecraft into the high bay area (see Figure C-17). Maneuvering room shall be provided to position the 30 L x 13 W x 13 H foot transporter. This loading dock will also be used to move MGSE into the high bay.

c. Structural

Requirements as defined in 3.2.1.

d. Mechanical

Requirements as defined in 3.2.1.

e. Electrical

(1) Exterior illumination shall be provided at 50 foot-candles to permit night off-loading operations.

(2) Communications shall be provided between lower crane operator and high bay crane operator to coordinate spacecraft hoisting operations.

(3) Spacecraft grounding during hoist operation is not required.

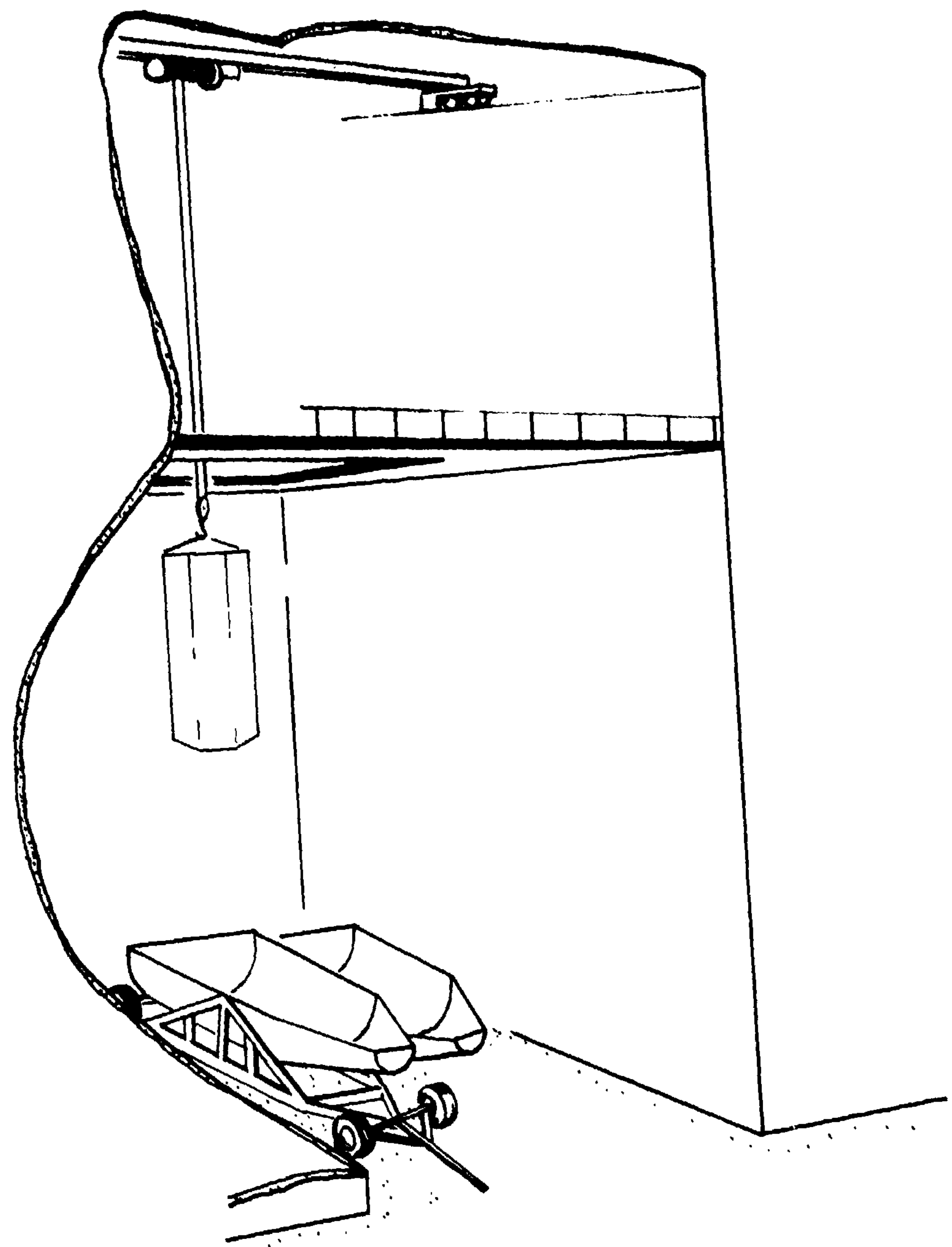


Figure C-17. Transporter receiving area.

3.2.4 Transportation Equipment

a. Civil.

Provide entry roads and sufficient room to maneuver a 12-ton flatbed trailer and tractor.

b. Architectural.

Provide warehouse type sheltered storage of at least 1500 square feet with a minimum of 16-foot ceiling height. Provide an entry ramp and a loading dock to permit ingress of spacecraft transporter (12-foot wide). Door size shall be at least 16 x 16 feet. Provide sheltered garage type storage for instrumentation trailer.

c. Structural.

Provide structure to support floor loads of empty transporter (30 x 12 foot base at 7 tons).

d. Mechanical.

Maintain a non-condensing environment.

e. Electrical.

General illumination at approximately 50 foot-candles.

3.2.5 Spares Storage Area

a. Architectural.

Provide a 500 sq. ft. room near the spacecraft assembly area. Partition the room with metal security screening to provide a 100 sq. ft. sub unit. Provide a padlockable door for the screened "bonded" stores area. Doors shall be standard 3 ft. x 6'8". Entry door shall be equipped with a cypher lock or equivalent.

b. Mechanical.

Provide a non-condensing environment controlled between 60-90°F.

c. Electrical.

Provide illumination greater than 50 ft.-candles and 100 ft.-candles at desk level. Provide telephone service.

3.2.6 Satellite Preparation Area

a. Architectural.

The S/C preparation area shall have a minimum of 2000 sq. ft. with full traveling crane coverage at a minimum hoist height of 35 feet above the floor level. The area shall be within 100 linear feet of the User Screen Room. The spacecraft (partially assembled), spacecraft MGSE and up to 21 bays of EGSE will be located in this area. Up to 18 people may occupy the area 24 hours a day for a given test sequence. Figure C-18 depicts a typical layout.

*Requirement based on partial assembly and checkout of spacecraft using peculiar EGSE (computer equipment is installed in screen room). Following assembly and check-out some of the EGSE will be moved into the screen room for the in-chamber tests. The Mark I preparation area meets these requirements.*

b. Structural.

Requirements as defined in 3.2.1.

c. Mechanical.

(1) General. Requirements as defined in 3.2.1.

(2) Ducted Air.
The area shall have two 8-inch diameter outlet ports for ducted air. Air temperature shall be adjustable between 65-75°F controlled to ± 3°F with relative humidity less than 50%. Air cleanliness sufficient to meet 3.2.1.2. Flow volume at bulkhead shall be adjustable between 1-125 lb/min.

*Requirements based on spot cooling needs of partially configured, powered spacecraft (partial thermal shrouding). Flow rates are derived from DSP and FLTSATCOM requirements.*

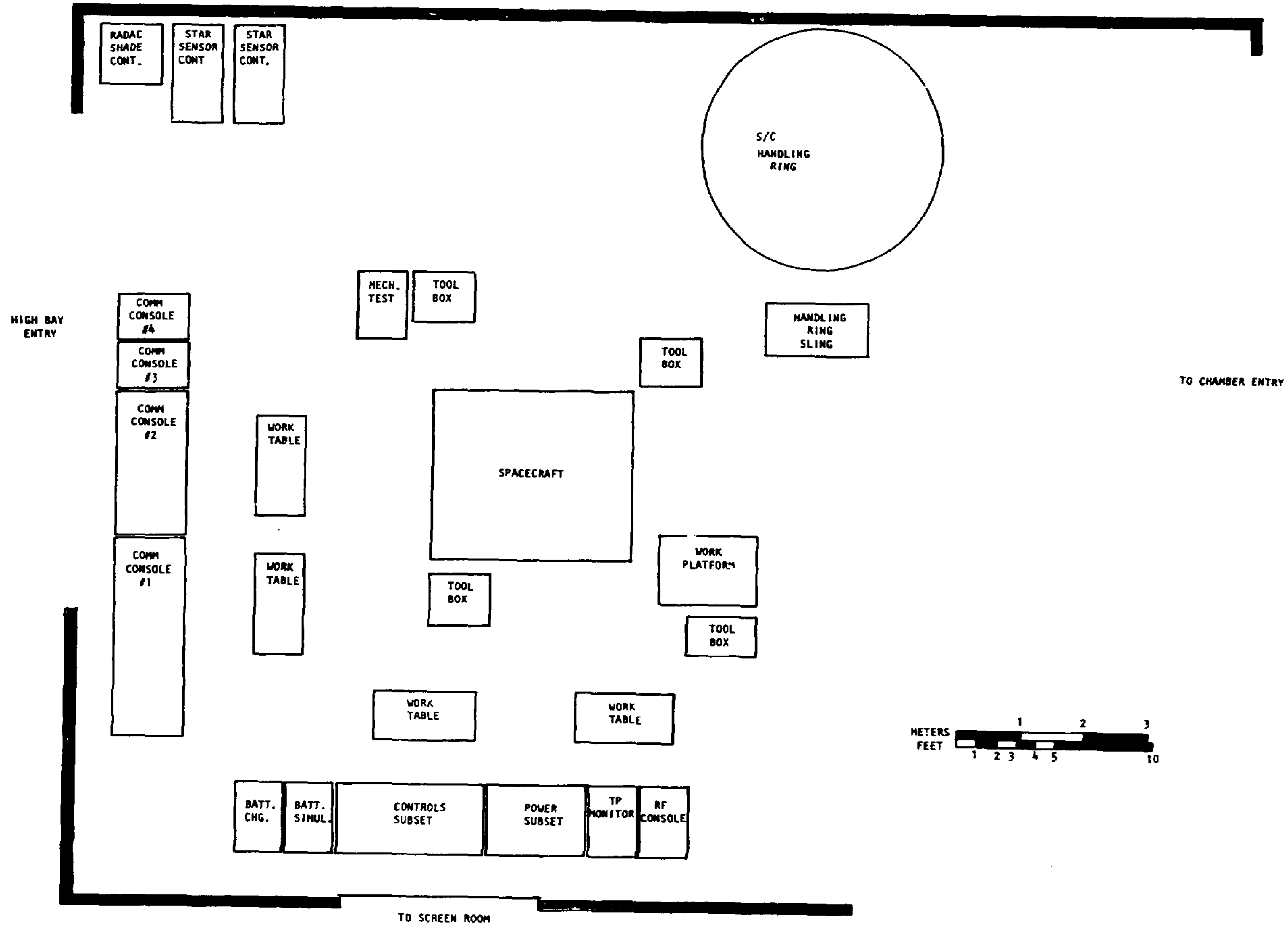



Figure C-18. Spacecraft preparation area example equipment installation.

d. Electrical.

(1) Power.

The following power shall be provided:

120 VAC, 60 Hz, 1 phase. 260A
208 VAC, 60 Hz, 3 phase, 60A

Detailed outlet locations will be provided in the user test plan.

(2) Illumination.

General illumination shall be a minimum of 100 foot-candles.

(3) Communications.

Provide dedicated three net communication between the EGSE area and the user screen room. Provide local telephone service in the area.

(4) Grounding.

Provide facility static grounds linking to the screen room central ground. Locate grounding points at central power entry and at two floor points in the maintenance area.

(5) Environment Monitoring Equipment.

Provide recording equipment to record temperature and humidity in the maintenance area.

(6) Cable Ducting.

Provide a shielded cable duct between the S/C preparation area and the user screen room. Minimum duct cross section shall be approximately 500 square inches.

*In general, cables leading from the preparation area to the screen room will be shielded. However, placing the cables in a shielded duct may serve to further reduce noise from facility sources which may be undergoing test while the S/C is being tested.*

3.2.7 User Screen Room

a. Architectural.

(1) The user screen room shall have a minimum of 1350 sq. ft. raised computer floor area. The room shall be located within 100 feet of the maintenance area and as close as possible to the chamber loading door. Up to 18 people may occupy the room 24 hours a day. Sanitary facilities should be nearby. Figure 3-19 presents a typical equipment layout.

(2) Ceiling height shall be 10 feet or greater above raised floor. Equipment entry doors shall be 8 x 8 feet clear area. Room shall have continuous RF shielding.

*Requirement on space related to 23 bays of computer equipment (FLTSATCOM) plus 29 bays of dedicated EGSE (DSP). This results in 52 bays of equipment at about 31 inches/bay. Front and rear access to 40 inch wide equipment results in clear space of 98 inches/bay. Thus the equipment requires 980 sq. ft. Table work space requires 5 x 25 sq. ft. or 125 sq. ft. Console operator positions require 14 x 5 sq. ft. or 70 sq. ft. Screen room penetrations will require 3 x 6 sq. ft. or 18 sq. ft. Minimum space would then be about 1200 sq. ft. Adding 10 percent for wasted area results in a total requirement of 1350 sq. ft. Space requirements assume all DSP EGSE would be installed in the screen room; the actual requirements should be somewhat smaller since some equipment may be locatable in the S/C preparation area.*

(3) Provide a break room with about 300 sq. ft. adjacent to the screen room. This room will be used by S/C checkout crew and screen room operators during breaks and shift changes. Maximum room load would be approximately 20 people.

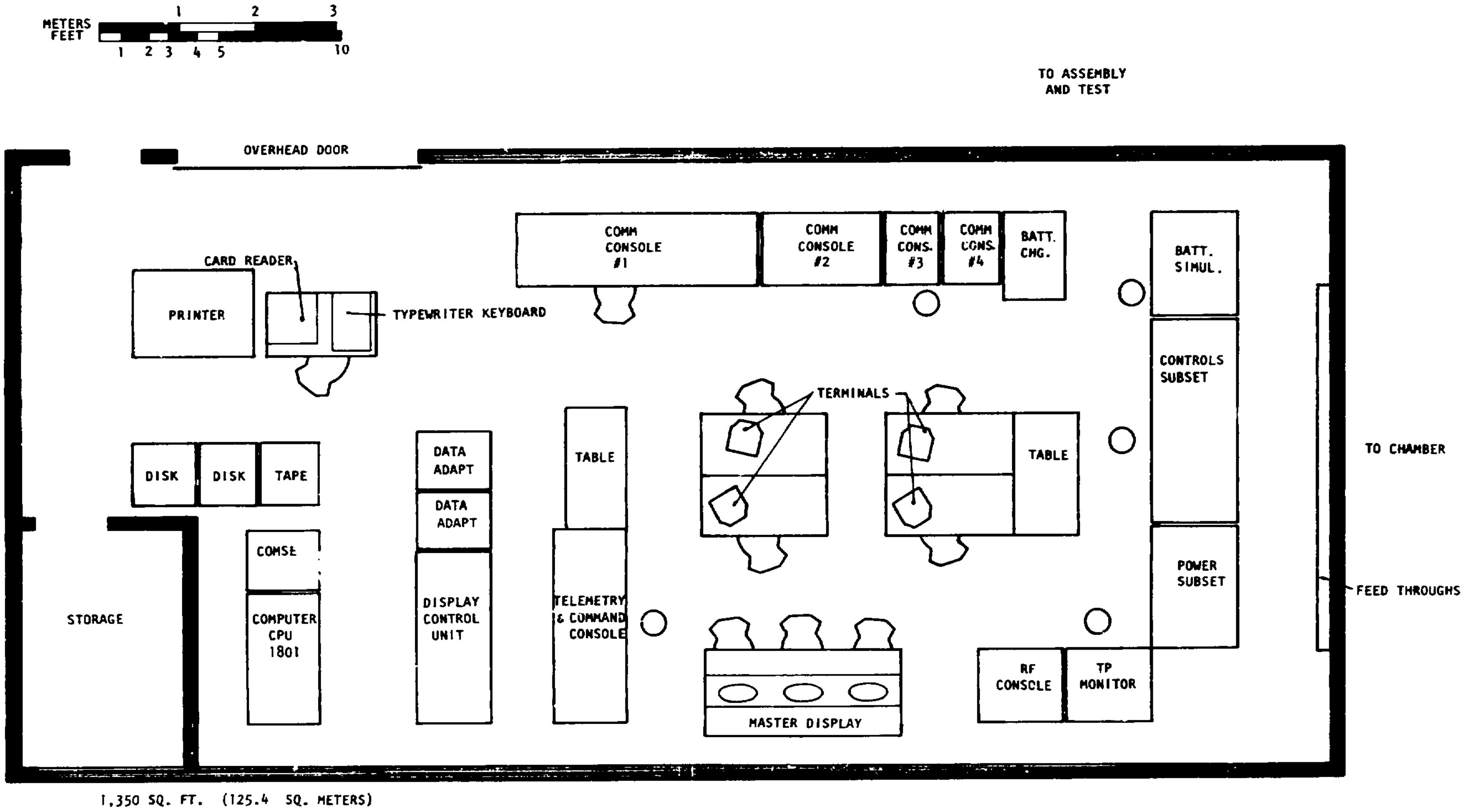



Figure C-19. User screen room typical installation.

b. Structural

(1) Floor Loading.
Floor shall be capable of 1000 PSI point loading. Total load of 52 equipment bays will be approximately 52,000 pounds loaded over 450 sq. ft.

*Requirement based on use of wheeled rack units with a typical weight of 1000 pounds per rack bay unit. Rack assemblies (consoles) can incoporate approximately 4 equipment bays resulting in a console length of about 8 feet and a weight of 4000 pounds loaded on four wheels (largest FLTSATCOM console).*

(2) Entry Ramp.
A ramp approximately 8 feet long shall be provided to transition between entry floor level and raised screen room flooring. Ramp shall not result in a decrease in the specified minimum screen room area.

*Requirement based on need to manually move 8 foot wheeled console through the door. Alternative methods such as forklift use are possible but could be difficult. The specified minimum screen room area (paragraph 3.2.7.a.(1)) did not include floor space provisions for an entry ramp.*

c. Mechanical

(1) Environment.
Under floor plenum shall provide conditioned and filtered air at 70 ± 5°F with relative humidity between 20 and 60 percent (non-condensing). Provide flow to hold temperature to 3.5°F/hour and 2% RH/hour rates of change.

*Requirement for under floor cooling air based on Digital Equipment Corporation computer facility recommendations. In general, most* *computers used for process control application require cooling air. Some high speed general purpose machines (IBM 370/168 and larger) require cooling water; such machines would not typically be used for spacecraft control.*

(2) <u>Acoustic Treatment</u>.
Provide acoustical treatment on walls and ceiling.

d. <u>Electrical</u>

(1) <u>Normal Power</u>. (Typical)
Provide the following power:

120 VAC, + 6%, -10%, 60 Hz ± 2%, 1 phase, 380A
208 VAC, + 6%, -10%, 60 Hz ± 2%, 1 phase, 15A
208 VAC, + 6%, -10%, 60 Hz ± 2%, 3 phase, 120A

Power delivery should be under floor and should be readily reconfigurable to serve different user requirements for outlet locations. Condition power to hold ± tolerances, be stable and noise free. Use isolation transformers to maintain screen room isolation.

*Power requirements were derived from FLTSATCOM total equipment requirements. Power tolerances are derived from Digital Equipment Corporation computer requirements; normal spacecraft EGSE permits ± 10% voltage variation and ± 5% frequency variation.*

(2) <u>Emergency Power</u>.
Provide emergency power to be available within 5 minutes after normal power shutdown:

120 VAC ± 10%, 60 Hz ± 5%, 1 phase, 380A
208 VAC ± 10%, 60 Hz ± 5%, 1 phase, 15A
208 VAC ± 10%, 60 Hz ± 5%, 3 phase, 120A

Deliver power through the same distribution system as the normal power distribution.

*Emergency power requirements permit full operation of all EGSE. Some equipment may not require operation under emergency conditions. Past experience, however, indicates that dual power distribution systems (normal and emergency) decrease reliability, require complex switching schemes and introduce a higher potential for human error. If a suitable emergency power scheme could be developed, the capacity requirement would be:*

*120A of 120 VAC, 15A of 208, 1 phase*
*60A of 208, 3 phase*

(3) Uninterruptable Power.

User will furnish uninterruptable power if required.

(4) Illumination.

Illumination intensity shall be 100 foot-candles at desk level.

(5) Communication.

Provide 3 channels of dedicated communication to the S/C preparation area and chamber interior. Provide standard five line telephone services to three telephones. Provide dedicated communication links between (1) the S/C test conductor and facility test conductor, (2) S/C thermal monitor and facility thermal control, (3) S/C test conductor and SGEMP data analyst. All stations should have selectable talk or listen capabilities.

(6) Grounding.

The grounding requirements of MIL-STD-1542 shall be used as a design guide.

(7) Central Timing.

A time of day display shall be provided based on central facility timing. IRIG-B parallel and serial timing shall be provided at TTL levels capable of driving 10 TTL loads. Countdown displays shall be provided for all facility automatic functions. Event markers from facility events shall be provided.

(8) Remote Monitors.
Remote displays of chamber wall temperatures and chamber vacuum status shall be provided.

(9) Video Displays.
Provide a television display for the chamber camera [3.2.8.d(3)]. Insure required video link does not compromise screen room integrity.

*Requirement is based on a need to avoid confusion about umbilical status during the test sequence. A video camera at the chamber [3.2.8.d(3)] provides an image of the umbilical connect area. The displayed image should provide a positive indication that the mechanical linkage has engaged. Corrective action could then be taken if electrical connectivity is not present.*

3.2.7.1 Cable Trays - Three cable trays connecting to the screen room shall be provided.

a. Preparation Area.
This cable tray shall connect the screen room to the S/C preparation area and shall be an RF shielded tray. Cross-sectional area shall be approximately 500 square inches. User will provide and install user peculiar cables. Screen room bulkhead connectors will be user furnished.

b. Chamber Entry.
This cable tray shall connect the screen room to the chamber entry door for temporary checkout cable installation. Shielding is not required; the tray shall be for physical protection. Cross-sectional area shall be approximately 200 square inches. Cable and screen room bulkhead connectors will be user furnished.

c. Chamber Umbilical

This RF shielded cable tray shall connect the screen room to a chamber bulkhead. Cables shall be provided for spacecraft umbilical power, electrical signals and fiber-optic signals. Interconnecting electrical cables and fiber-optics shall be designed for multiple, general purpose use.

Typical cable configuration may be as follows:

| | |
|---|---|
| Spacecraft Power | 32VDC, 60A maximum |
| Spacecraft Power Return | 32VDC, 60A |
| 10 Twisted shielded pairs | 28VDC, 2A max |
| Three sets of cable spares | |

Cable shall connect to SXTF provided bulkhead feedthrough connectors at the screen room disconnect and the chamber umbilical feedthrough.

d. Chamber Spacecraft Functional

A shielded cable tray shall connect the RF probes in the chamber to the user screen room. Cross-sectional area shall be approximately 200 square inches. Typical conductors could consist of two S-band waveguides and ten low loss 50Ω coaxial cables.

3.2.7.2 Screen Room Disconnect

a. User Provided Cables.

During testing, user will provide filtering for or disconnect user provided cables at screen room interior. Feedthrough connections will be capped if disconnected.

b. Chamber Interface Cables.

During testing, facility provided umbilical interface cables shall not conduct radiated energy into the screen room. This may be done by adequately shielding the cables by filtering the inputs or by automatically disconnecting the cable from the screen room and capping the penetrations. If used, disconnect operation shall interlock pulser controls and shall provide annunciation signals to user screen room and to facility control. Filtering, shielding or disconnect is required to prevent pulser RFI or test SGEMP from upsetting screen room equipment.

3.2.7.3 Umbilical Interface - This screen room interface is described in 3.2.7.1c and 3.2.7.2b. The SXTF shall provide connectors for the chamber-screen room bulkhead feedthrough connectors. A 22 pin connector for the indicated lines of paragraph 3.2.6.1c should be adequate. The additional three spare connectors shall be capped using SXTF furnished caps during testing.

3.2.7.4 Fiber Optic Interface - The SXTF shall provide interfaces for the fiber-optic cables connecting to chamber feedthroughs and terminating inside the screen room. All fiber-optic penetrations shall be RF ducted to reduce RFI. General purpose facility provided fiber-optic transmitter and receiver units shall be available for S/C installation and compatible receiver/transmitter units shall be installed in the screen room. Approximately 16 lines of fiber optic cable should be provided. Detailed link requirements will be identified in a separate analysis.

3.2.7.5 Dielectric Waveguide Interface - The user will provide standard waveguide to dielectric transitions. Facility shall provide dielectric waveguides suitable for transmission of RF between 2 and 12 GHz. The dielectric waveguides shall be available to the user at chamber center and at the screen room feedthrough. Sufficient slack in the chamber waveguide shall be provided to permit S/C rotation without entanglement of the waveguide with the suspension lines.

3.2.8 Test Chamber Area

a. Architectural.

Chamber shall have a platform floor level with lower exit door to permit entry and exit of MGSE.

b. Structural.

Requirements as described in 3.1.2. Platform Floor Loading shall permit a total load of 7.5 tons and shall permit fastening of support fixtures to chamber platform floor.

*Total load requirement based on the following estimated weights: (1) spacecraft - 6000 pounds, spacecraft holding fixture - 2000 pounds, solar panel and attachment structure - 2000 pounds, panel holding fixtures - 1000 pounds, fiber optic, dielectric suspension and dielectric strongback - 2000 pounds, and miscellaneous MGSE (work stands, alignment equipment, etc.) - 2000 pounds. The spacecraft holding fixture must be fastened to the floor in order to handle the cantilever moment.*

c. Mechanical

(1) General.

Requirements as described in 3.1.2.

(2) Ducted Air.

Requirements as described in 3.2.6.d.(2).

d. Electrical

(1) Illumination.

Provide general illumination with a minimum of 100 foot-candles at the chamber center from the floor to halfway up the chamber. Illumination shall not interfere with crane operators' (upper or lower) ability to accurately operate crane.

(2) Communication.

Provide dedicated communications between lower chamber, S/C preparation area, and screen room.

(3) Television.

Provide visual monitoring coverage remoted to the user screen room for monitoring umbilical retraction/insertion sequences and for monitoring spacecraft movement between test sequences.

3.2.8.1. Suspension System - The suspension system provides the mechanical interface between the spacecraft and the chamber. The suspension system includes a rotary traveling fixture, a traversing fixture and a spacecraft suspension strongback. The AEDC loading procedure (See Appendix A, this report) indicates that the suspension system should maintain the chamber upper entry door aperture during chamber loading. The crane will be used to lower the strongback suspension system to the S/C for attachment. Once attached to the S/C the strongback will be raised to top of chamber and coupled to the chamber rails. At this point the crane will be removed and the chamber lid installed. The transfer procedure obviates the need for hoisting by the traversing fixture. Positioning requirements of the strongback suspension system are:

| | |
|---|---|
| Translation - | |
| Maximum rate | 1 ft/min |
| Maximum acceleration | 0.5 g |
| Maximum jerk | 0.5 g/sec |
| Accuracy | 6 inches |
| | |
| Rotation - | |
| Maximum rate | 3°/min |
| Maximum acceleration | $0.125°/sec^2$ |
| Maximum jerk | $0.125°/sec^3$ |
| Accuracy | 1° |

The strongback shall be a facility provided fixture. The strongback will be basically a dielectric material which will attach to a support structure and ring mounted within the chamber at or near the top opening of the chamber. The support structure shall not restrict the 20 ft. diameter opening of the chamber. The user will provide dielectric suspension lines (fillistrand or equivalent) to suspend and attach the S/C to the strongback. The strongback and support structure shall be capable of supporting 8,000 lbs. minimum.

*Suspension requirements are based on the preliminary conclusion that support from above will be the preferred method. Further studies may be appropriate to verify this conclusion. The reason that the overhead suspension appears to be preferrable is that it minimizes the amount and complexity of extraneous material in the test volume. Even dielectric material can interfere with the photon experiment and will certainly interfere with spacecraft charging experiments. The amount of material to support the test object weight as well as the fixture's own weight will not be insignificant. Furthermore, a structure capable of supporting the spacecraft from below will have to have a large enough base to prevent toppling.*

*Considering spacecraft like FLTSATCOM and DSCS-III, the fixture would have to have a large open area directly below the center body for the lower solar panel and a lattice work above the spacecraft for the upper solar panel. It was because of these configurations that we reached our preliminary conclusion.*

*The suspension requirement of 8000 pounds was based on an estimate of up to 6000 pounds for a spacecraft and 2000 pounds of additional support and strongback equipment. The spacecraft weight is an estimate based on growth from current generation spacecraft to the limit of spacecraft that can be tested in the AEDC chamber. For example the dry weight of FLTSATCOM is about 2000 pounds as is the current generation DSP. The upgraded DSP is about 4000 pounds and the HEAO which has a very large center body (12 feet diameter, 20 feet long) representative of what might be a very large test candidate weighs 6000 pounds.*

3.2.8.2 Fiber Optic (F/O) Interface

Data F/O - Data fiber optic cables include transmit cables from spacecraft and receive from facility control cables. In order to reduce size of F/O transmitters, each transmitter should be controlled through a separate F/O line. For 20 F/O data lines, there will be 20 F/O control lines; thus, the facility should provide chamber penetrations for 40 F/O (reference JAYCOR RE-79-2066-129, 200-80-215/2066 and HDL Briefing Notes, August, 1980). Penetrations for control lines should be separate from data lines. Control lines will be low data rate, high tolerance links which can use numerous connections without serious link degradation. Data lines will be high bandwidth lines, potentially single mode (small) fibers, which may not use connectors at the chamber penetration (hard seal). Data lines may require replacement after several spacecraft tests. Data F/O links run from the spacecraft, through chamber penetration(s), to the facility screen room.

Spacecraft Control F/O Links - Facility provided F/O receiver-transmitter units shall be compatible with receiver-transmitter units of Paragraph 3.2.7.4. Sixteen F/O penetrations through the chamber to the screen room shall be provided.

3.2.8.3 Umbilical Interface - The chamber shall accommodate penetrations required by 3.2.7.3. The connector penetrations shall be replicated at both the top and bottom of the chamber, permitting either upper or lower umbilical installation. Interface from the chamber wall to the spacecraft will be the responsibility of the facility user. The facility shall provide mating connectors for the chamber interior interconnect.

3.2.8.4 Chamber Thermal Requirements - The chamber shall provide the capability to maintain active spacecraft electronics within safe operating temperature limits. This can be accomplished by at least two methods which are acceptable from thermal environment point of view. One method is to use a liquid nitrogen ($LN_2$) thermal shroud. The thermal shroud shall have a emissivity greater than 0.8 and shall cover greater than 4.4 steradians of solid angle viewed by a spacecraft panel at the center of the chamber facing any direction (except toward the photon source). Eight independently controllable zones are required if an $LN_2$ shroud is used.

A preferred approach is the use of a gaseous nitrogen ($GN_2$) thermal shroud with temperature controllable from -200 to +100°F. Six to eight independently controllable zones are preferred but not absolutely required.

The MBS and PRS are to be covered with a thermal shield. The basic shield should have an emissivity greater than 0.8 and be convex with respect to the inside of the chamber. The shield should have an emissivity of less than 0.2 on the side facing the sources.

*The present AEDC configuration will meet the first requirement described above. The 4.4 steradian requirement is based on 70% cold wall coverage of the hemisphere (2π steradians) viewed by a planar spacecraft panel of any size located near the center of the chamber. This requirement can be met without having thermal panels on the floor or ceiling.*

*The extant ability to operate each of the eight side wall panels independently gives intermediate temperature capability so that $GN_2$ shroud operation is not mandatory. However, without $GN_2$ capability special heaters may be required on the spacecraft to prevent certain areas from dropping below acceptable temperature limits.*

*The source cover with sufficient convexity allows the spacecraft to "see" a reflection of the thermal shroud in the source cover. Therefore, unless the spacecraft panel is right next to the source, it will effectively be cooled by the shroud in other parts of the chamber.*

3.2.8.7 *Work Platforms* - The facility shall provide work platforms to permit work on the suspended spacecraft. Removable platforms shall be removable through the lower chamber entry. Installed platforms shall not restrict S/C motion or degrade cold wall/radiation shield performance.

3.2.9 *MGSE Storage Area*.

a. *Architectural*.

MGSE storage shall be provided adjacent to the S/C preparation area and lower chamber entry area. Storage near the S/C preparation area shall be approximately 2,000 square feet. Storage at lower chamber area shall accommodate equipment such as work platforms and spacecraft supporting fixtures and shall be approximately 200 square feet. MGSE lowered into the chamber will be capable of passage through the 8 foot lower opening.

b. *Structural*.

Requirements as defined in 3.1.2.

c. *Electrical*.

Illumination in storage areas shall be a minimum of 50 foot-candles.

# APPENDIX D

# ACRONYM LIST

## ACRONYM LIST

| | |
|---|---|
| ACTE | Automated Communications Test Equipment |
| ACQ | Acquisition |
| AEDC | Arnold Engineering Development Center |
| AI&T | Assembly Integration & Test |
| APT | Automatic Pointing and Tracking |
| APU | Auxiliary Power Unit |
| ASAT | Anti-Satellite |
| ATM | Assistant Test Manager |
| | |
| BHO | Beam Handling Optics |
| | |
| CDR | Critical Design Review |
| $CO_2$ | Carbon Dioxide |
| CO | Carbon Monoxide |
| COMINT | Communications Intelligence |
| COMMS | Communications Security |
| COMSEC | Communication Security |
| CM | Countermeasures |
| $CM^2$ | Square Centimeters |
| CPU | Central Processor Unit |
| CRT | Cathode Ray Tube (Video Display) |
| | |
| DF | Deuterium Fluoride |
| DNA | Defense Nuclear Agency |
| DSP | Defense Support Program |
| | |
| ECCM | Electronic Counter-Countermeasures |
| ECM | Electronic Countermeasures |
| EGSE | Electronic Ground Support Equipment |
| EM | Electromagnetic |
| EMP | Electromagnetic Pulse |
| ERP | Effective Radiated Power |
| ETS | Electrical Test Supervisor |
| EW | Electronic Warfare |

| | |
|---|---|
| Fb | Frequency Demultiplexor (Encryter Key Selector) |
| FB GOE | (Encrypter) Fleet Broadcast |
| FDR | Final Design Review |
| FLTSATCOM | Fleet Satellite Communications |
| F/O | Fiber Optics |
| FSC | Fleet Satellite Communications |
| GHz | GigaHertz |
| $GN_2$ | Gaseous Nitrogen |
| GSE | Ground Support Equipment |
| HDL | Harry Diamond Laboratories |
| HEL | High Energy Laser |
| HF | Hydrogen Fluoride |
| IFJ | In-flight Jumper |
| I/O | Input/Output |
| IPL | Integration Planning and Logistics |
| IRIG-B | Interrange Instrumentation Group, Standard Timing Format B |
| IWETF | Integrated Weapons Effects Test Facility |
| J/S | Jammer-to-Signal |
| JSC | Johnson Space Center (Houston) |
| Kx | Encrypter Key |
| $LN_2$ | Liquid Nitrogen |
| MAGE | Mechanical Aerospace Ground Equipment (MGSE) |
| MBS | Modular Bremsstrahlung Source |
| MGSE | Mechanical Ground Support Equipment |
| MTS | Mechanical Test Supervisor |

| | |
|---|---|
| NASA | National Aeronautics and Space Administration |
| | |
| OCXO | Oven Controlled Crystal Oscillator |
| | |
| PDR | Preliminary Design Review |
| PRS | Plasma Radiator Source |
| PSI | Pounds per Square Inch |
| PT | Pointing |
| | |
| QA | Quality Assurance |
| QAM | Quality Assurance Manager |
| | |
| RF | Radio Frequency |
| RFI | Radio Frequency Interference |
| RH | Relative Humidity |
| Rx | Receiver |
| | |
| SA | Solar Albedo |
| SAT | Satellite |
| S/C | Spacecraft |
| SCFH | Standard Cubic Feet per Hour |
| SGEMP | System Generated Electromagnetic Pulse |
| SHF | Super High Frequency |
| SIGINT | Signal Intelligence |
| SM | Satellite Manufacturer |
| SNR | Signal-to-Noise Ratio |
| SPADOC | Space Defense Operations Center |
| SXTF | Satellite X-ray Testing Facility |
| | |
| TA | Threat Avoidance |
| TD | Test Director |
| TEMPEST | Project Name for Compromising Program |
| TM | Test Manager |
| TP | Test Point |
| TT | Threat Tolerance |

| | |
|---|---|
| TT&C | Telemetry Tracking & Command |
| TTL | Transistor - Transistor Logic |
| Tx | Transmitter |
| VAC | Volts Alternating Current |
| VDC | Volts Direct Current |
| VHF | Very High Frequency |
| $W/CM^2$ | Watts per Square Centimeters |

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